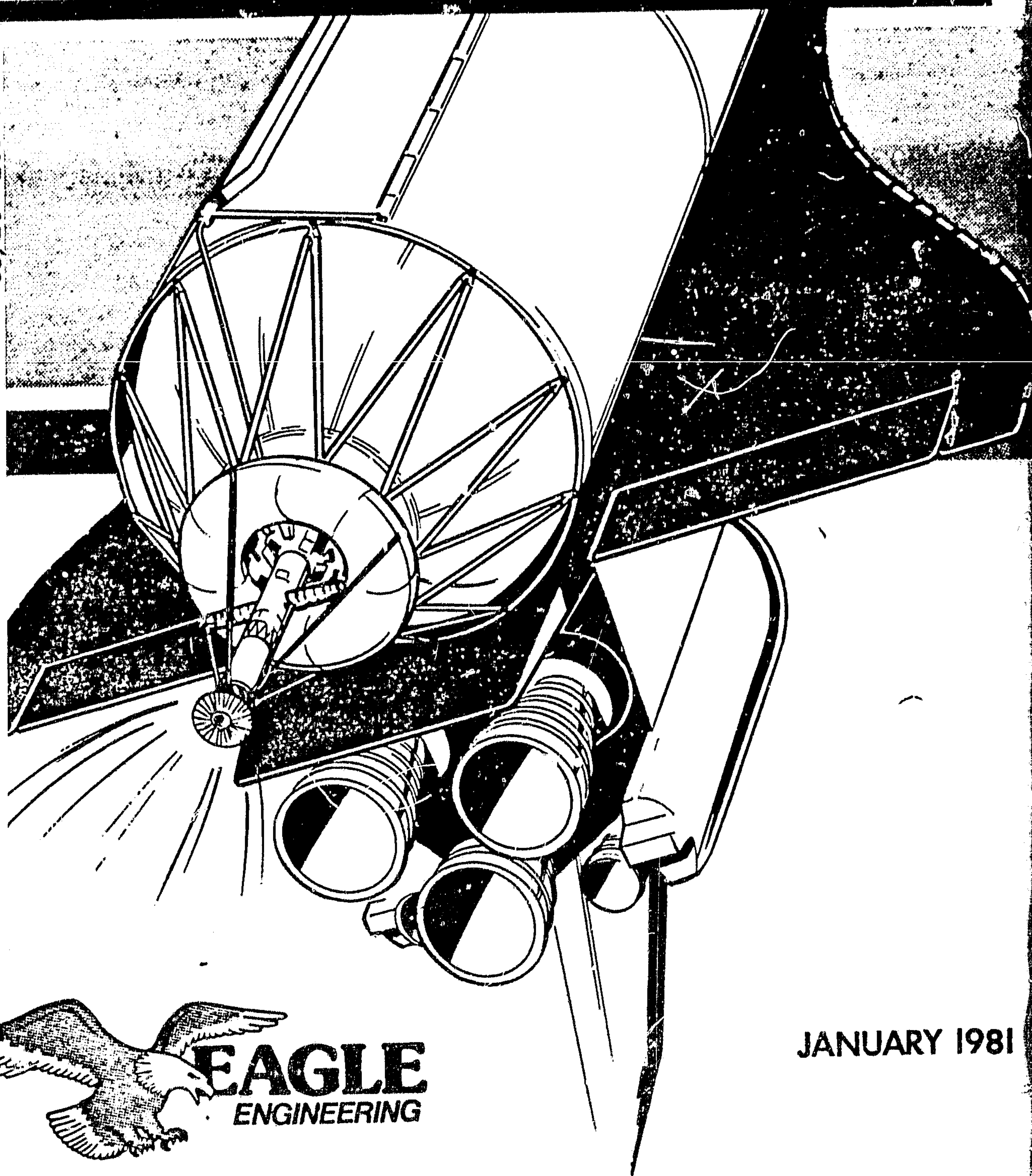
MPD Thruster Application Study

JPL R&D CONTRACT
NO. 955842

(NASA-CR-165481) MPD THRUSTER APPLICATION STUDY (Eagle Engineering, Inc.) 38 p
HC A03/MF A01 CSCL 21C
N81-27193
Unclas
G3/20 26917



JANUARY 1981

EAGLE
ENGINEERING

PREFACE

The NASA Jet Propulsion Laboratory (JPL) is developing the Magneto-Plasma-Dynamic (MPD) thruster, an electrically-powered rocket engine for future space flight use. Eagle Engineering, Inc. of Houston, Texas, was awarded a contract to assist in development planning and application of the MPD. This work was accomplished between July, 1980 and January, 1981, and the results of this work are presented in this report. The study director at JPL was Dr. Kevin Rudolph. The Eagle Engineering study team was:

Hubert P. Davis, P.E.	-	Study Manager
Rudy Williams	-	Thermodynamics
William Gill	-	Structures Analysis
William Stump	-	Research Assistant

TABLE OF CONTENTS

JPL - MPD

Magneto - Plasma-Dynamic(MPD) Thruster Application Study

1. Introduction

The Jet Propulsion Laboratory of NASA, in concert with the USAF Rocket Propulsion Laboratory, is currently conducting exploratory development of an electrically - powered rocket engine for future space transportation application. Significant research has been completed on the Magneto-Plasma Dynamic (MPD) thruster at Princeton University which suggests that this engine concept may, for some applications, become competitive with or superior to the ion bambardment engine. The latter engine is currently in an advanced state of development and is considered "technology ready" for application to a Solar Electric Propulsion Stage (SEPS) in the 25 to 50 Kwe class.

Eagle Engineering, Inc., was granted JPL R & D Contract 955842 in July, 1980 to assist in development planning and application studies for the MPD thruster. This report provides the data generated under that contract, supplementing the presentation of results tendered to JPL on October 28, 1980 at the semi-annual MPD program review held at Princeton University. Charts used for this presentation are attached as Appendix A to this report.

2. Application Studies

The ion engine displays outstanding performance characteristics for extremely energetic planetary missions, such as the proposed Halley Comet rendevous. Its use of mercury propellant, complex power conditioning requirements and inherently low thrust per unit (0.37 N and 0.13 N/Kwe for the

30 cm diameter engine) may limit its utility for certain future mission requirements which are just now being defined. Both scarcity of the commodity and potential deleterious effects upon the earth may lead to a change of propellant from mercury to a cryogenically stored gas such as argon. Although this change offers approximately double the specific impulse from 3,000 seconds to 6,000 seconds, it also halves the thrust produced per unit of electrical power supplied. For Earth orbit - raising application, in particular, acceptable trip times through the trapped radiation belts may require many thrusters and high electrical power supply.

Optimization studies have indicated that, unlike conventional chemical rocket engines, increased specific impulse may not be advantageous from a systems viewpoint. For low thrust orbit raising of large space structures, a velocity change of about 6 Km/Sec. is required to acquire the geo- stationary orbit . At 6,000 sec. specific impulse, the required propellant is about 10.7% of final geo-stationary orbit mass. At 2,000 sec. specific impulse, the required propellant increases to about 35.8% of final mass, an increase of about 23% in the mass which must be placed into low Earth orbit. The relation for electric thrusters which governs thrust output is:

$$F = \frac{2 \cdot \eta P}{I_{sp} g_o}$$

F = thrust
η = thrust efficiency
P = electrical power supplied
I_{sp} = specific impulse
g_o = gravitational constant

Thus, if trip time requirements dictate the acceleration level which is needed, as specific impulse is increased, the

power supply must increase in size and the thrust must be increased further to drive the increased mass of the power supply. The result is that, given comparable thrust efficiency, a device producing about 2,000 seconds specific impulse can provide a lower initial mass in low orbit, a smaller power supply and thus a more economical transit from low to nigh earth orbit than the apparently higher performance 6,000 second unit.

Chart 3 of Appendix A illustrates two important facets of the large space structure transfer problem. First, for space structures up to 20,000M^2 in area (about 160M diameter if circular), there is no significant reduction in structural mass by reducing transfer acceleration levels below 1×10^{-3}g. Second, the scales at the top and bottom of the chart illustrate the penalty in trip time associated with lower acceleration levels. This chart was intended to make the point that low thrust chemical rockets, providing an acceleration level of about 1×10^{-2}g and a 3-day trip time are preferable to electric (ion) propulsion producing about 4×10^{-5}g and about 200 days trip time and that the IUS and Centaur (even at tank-head idle thrust) are not suited for this mission. Responding to these data, the NASA has active on-going studies of the low thrust chemical transfer vehicle and engines for application in the late 1980's or early 1990's. Propellant required is about 2 to 3 times final mass in geo-stationary orbit.

An equally valid point is that an inherently lower specific impulse electric thruster is desirable - the MPD is such a device. The MPD stage potentially offers a reduction in launch mass by a factor of over 2, compared to chemical propulsion stages and delivers, with the payload, a large electrical power supply which may be utilized to serve the payload requirement for its operational life. No definition studies are currently underway on an MPD stage.

It became evident, in the NASA/DOE studies of 1975-1979 of the Solar Power Satellite (SPS) orbit transfer, that electric propulsion was a necessity if transportation costs (dominated by launch vehicle costs) were to remain within acceptable bounds. Numerous studies were performed both in-house (JSC and MSFC) and under contract (Boeing and Rockwell) with both "self-powered" (a part of the 1 to 4GWe SPS Module array was partly deployed for propulsion power) and "independent" flight modes considered. The SPS studies agreed, in their later stages, to baseline the "independent" flight mode and the "independent orbit transfer vehicle." (IOTV) was conceptually defined. Chart 4 of Appendix A illustrates the IOTV selected by Boeing for transfer of SPS components and logistics support from low orbit to Geo.-stationary orbit where the SPS's were to be built. The thrust-to-weight ratio of the stage varied from about 6×10^{-5}g at ignition to about 2×10^{-4}g upon arrival at the geo-stationary orbit, with a trip time of about 6 months.

The long trip time had several undesirable features. First, the expensive SPS program inventory was tied up in non-productive transit for half a year (something to consider with the prime interest rate at 20%). Second, the solar array remained for extended periods in the intense radiation field of the trapped radiation belts. The electrical output of the array is severly degraded by this passage - with loss of output estimated to be 20% to 50% per passage, according to one observer (Chart 5 of Appendix A).

For a reusable IOTV, it was concluded by the SPS studies that in-space laser annealling would be employed between missions to restore array output. A far more satisfying solution would have been to employ the MPD thruster, decrease the trip time with a smaller (and less costly) solar array and accept the 10 to 20% loss in output per trip. Unfortunately, the MPD thruster had not yet been characterized sufficiently to commit the SPS "reference system" to any but the ion - bombardment thruster.

A number of important operational factors of electric propulsion orbit - raising were also tentatively addressed in the SPS studies. The impact of these factors upon stage design and its propulsion system requirements were less clearly defined. Charts 6 and 7 specify some of these operational factors. From a 28½^{0} inclined low orbit, the degree of solar occulation experienced during transit depends upon solar day of departure and how quickly the vehicle climbs into unobscured sunlight. During each occulation, array power is lost and chemical propulsion must be employed to maintain the desired vehicle

altitude - the amount needed is doubly dependent upon vehicle acceleration as both the number of occulations and their average duration depend upon the available acceleration rate. Each time the vehicle exits from the Earth's shadow, the solar array experiences the insolation transient and the electric propulsion engines are re-started. Start-stop response time and performance loss are not yet fully understood but are clearly a factor to be minimized in order to preserve high mission-duty-cycle average performance and to achieve long-lived engine components.

Although not yet a critical problem, the potential for collision of large space structures with orbital debris is of increasing concern and interest. Chart 8 (Appendix A) illustrates the estimated collision potential for the shuttle orbiter with debris population estimates as of 1978. These data do not cause serious concern for safety of the relatively small shuttle orbiter, partly because most missions will remain below 300 Km altitude and are relatively brief.

For a large structure (20,000M^2 class), requiring a half year or more for the transit, operating in an environment which is certain to be worse than it was in 1978, the potential for collision may be sufficiently high to warrant debris detection/evasion capability. Clearly, limiting the time of exposure to tens of days is preferable to exposure for hundreds of days. Equally clear, the primary propulsion

system acceleration capability will determine required debris tracking range for a detection system.

Also related to trip time are the potential for system failure, either of the transfer vehicle or its payload and costs for mission operations support.

The conclusions reached from these application studies are summarized on Chart 9. The fundamental conclusion is that large space structure transfer needs electric propulsion at moderate specific impulse to reduce launch costs and that required acceleration rates call for a thruster having the characteristics now expected of the MPD.

An example of an "applications study" is illustrated by Appendix B. The problem posed in this instance was to compare the mass of a solar array interconnected to supply 240VDC for "direct drive" of a continuously - operated MPD thruster with 2400VDC solar array collection and power conditioning to the required 240VDC operating level. Although the data utilized in this study may not be representative of best available technology in either case, the indications are that high voltage array power gathering and power conditioning is the preferred approach. More study of the space plasma/MPD exhaust plume/solar array potential is needed to predict power losses due to plasma shorting at various altitudes of the transit. Earlier work by JSC indicated that, although geo-stationary orbit power collection at 40KV was desired for operation of a 5 to 10 GWe SPS, voltage during transit would be limited to about 2400V to avoid excessive plasma shorting power loss.

The point made by this discussion is that the MPD engine, its spacecraft and the natural environment interact in ways which can only be determined by detailed and comprehensive application, or systems, studies of MPD-powered flight vehicles. These studies can be expected to result in altered perception of desirable MPD flight engine characteristics and of the MPD technology development program.

Charts 9 and 10 of Appendix A summarize the findings of this brief review of MPD flight application issues. First, transfer of large space structures from low orbit to the geostationary orbit can benefit greatly from the specific impulse provided by electric propulsion thrust devices. Most of the benefits are, however, acquired at modest specific impulse level in the vicinity of 2,000 seconds. Second, the transfer maneuver should be completed in less than 100 days, for a variety of reasons, therefore vehicle acceleration in the 10^{-4}g range is necessary. Third, the ion bombardment engine operating with gaseous propellant for orbit transfer, may not enjoy the high efficiency levels predicted for 10^{-5}g range of vehicle acceleration which is adequate for many planetary missions. Fourth, and finally; before application studies may be conducted with confidence, predictions are needed of the MPD flight engine characteristics. Characterization of both the thruster and power conditioning is needed for both steady-state and pulsed operation over a range of thruster sizes requiring from less than 1 to about 10MWe, operated over a band of power level from full power (near J*)

to less than half rated power. Full exploitation needs to be made of the current analytic understanding of the MPD and of the rapidly growing body of laboratory test data to produce the needed "performance maps" of an MPD thruster family.

3. MPD Flight Engine Considerations

In order to be a serious contender for flight vehicle application, an MPD thruster must meet criteria which are mission-peculiar.

For example, an MPD powered vehicle used for low orbit to GEO-stationary orbit raising with no return of the system may need an operational lifetime of less than 1,000 hours. Planetary missions may require longer endurance. A reusable electric OTV may need to fly 10 or more round trip missions inferring thruster lifetime of 20,000 hours or more. An engine utilized for a planetary mission may be powered essentially continuously, given the minimum altitude which may be required for nuclear reactor startup, whereas an engine used for orbit - raising may be required to tolerate hundreds of start - stop cycles due to occulation of the solar array.

Similarly, in order to successfully compete with the ion bombardment and chemical propulsion alternatives, the MPD engine must produce a thrust efficiency which is mission - dependent. These parameters can only be determined by application studies, but are important to the setting of research goals and of evaluating progress.

Chart 11 of Appendix A lists requirements of a flight engine which are considered to be identifiable today but have not yet been treated in the available literature on the MPD.

Engine cooling will be necessary if the compact, high power MPD engine is to survive for the required mission life. Quasi-steady state testing with infrequent short pulses of energy have indicated some erosion. The research thruster parts operated as heat sink units and thermal degradation was almost certainly due to surface rather than bulk heating. Research data indicates that the cathode (center body) of the MPD is protected by a plasma sheath and an outward flow of electrons. Estimates have been made at Princeton that about 10% of the input power is deposited to the annular anode, leading to significant localized heating and ablation of the tungsten anode of the research engines. As more efficiency data becomes available from quasi-steady state laboratory testing, an energy budget is needed to better quantify and locate the several contributors to power loss. The basic feasibility of steady-state MPD operation hinges upon the ability to maintain acceptable equilibrium temperature of available engineering materials. Engine configuration, active thermal control and material studies are needed in order to determine feasibility. These studies should be based upon more knowledge of the power losses and location of deposition zones of the wasted power.

In addition to the primary electrical - conducting parts, radiative heating of other engine parts may be expected. For steady-state operation in particular, the

injector face and thrust chamber insulating material may require active cooling beyond that provided by propellant flow. In addition, electrically - operated propellant valves in other physically small engines have required thermal isolation to prevent thermal failures.

The electrical power supplied to the MPD is at a sufficiently high power level and low voltage such that tens of kilo-amperes must be routed from power conditioning units to the thruster. For ground - based exploratory testing, massive buss bars are an adequate and appropriate solution For a flight engine, however, close coupling of power conditioning and the thrusters will become necessary to mimimize weight. It is possible that direct mounting of an MPD thruster to its power conditioner may become attractive. Candidate flight engine designs and power distribution studies are needed to define packaging of both the thruster and power conditioning apparatus.

Thrust vector control is an operational necessity; gimballing of engines has been the traditional means of providing pitch and yaw control, with paired operation of primary or secondary engines providing roll control. Given the inherent high power consumption of MPD thrusters, only a single engine may be able to obtain power from the available power supply In this instance, either a capable attitude control system or MPD engine gimballing must be provided. Location of the thrusters must be chosen to avoid impingement of the MPD exhaust plume upon space vehicle structure. Again, vehicle design studies are needed, based upon good exhaust plume

characterization over the range of power and propellant flow needed.

Given the compactness and potential light weight of the MPD thruster, it may prove advantageous to provide flight spare engines on a vehicle, either switched on as needed or utilized alternately to reduce the active cooling needed. Cross-strapping of thrusters and power conditioners is desirable to achieve desired system reliability. This feature may be difficult to achieve if close - coupling is used to reduce electrical conductor mass. System definition and fault-tree analysis will be necessary in order to establish reliability goals for the MPD propulsion system.

Figure 12 of Appendix A illustrates one possible physical arrangement of a steady - state MPD thruster for flight vehicle use. Supplemental radiation cooling is illustrated, using heat pipes to aid in waste energy rejection to space. The "first look" estimate of heat load for this thruster indicates that this design may require a four-fold increase in acceptable heat pipe evaporator energy density over current heat pipe state-of-the-art.

4. Thruster Ground Testing:

Early in this study, a study team member was asked to review the MPD literature then available and to prepare his observations bearing on the MPD ground test program. These observations are provided as Appendix C to the report. A salient point is that, given ground test limitations, different test facility and research engine designs may be necessary to

acquire data on the separate areas of interest in MPD exploratory development.

Another team member determined the characteristics of 31 large thermal - vacuum and vacuum chambers in the United States. These data are included as Appendix D. No detailed assessment of their suitability for MPD testing was performed. Particularly lacking at this time is treatment of the utility of these metal - walled chambers for test of the plasma - producing MPD thruster units. Pumping capacity was reviewed which indicates that a 2% to 4% duty cycle at quasi - steady state is the most which can be expected in available facilities, with pulse frequency and width to be determined by flow rate and chamber size. Additionally, removal of the large thermal load produced by the MPD may pose a host of new problems for cryogenically - cooled wall vacuum maintenance systems.

In summary, the construction of dedicated, di-electric wall MPD vacuum chambers at Princeton and JPL is considered to have been an appropriate step for exploratory development. Further study is needed of facility requirements for flight engine development and ground qualification testing. It is considered likely that a new major test facility investment will become a necessity to fully develop the MPD thruster concept. Limitations of ground testing will always be severe, however, and a trade study needs to be conducted to determine a cost-effective mix of ground and space flight testing.

5. Thruster Flight Testing

Given the 5 MWe power use and 6 gram/sec. plasma flow rate of the MPD thruster, early flight testing in the space environment will probably become necessary. The space

shuttle vehicle, properly equipped, may be ideally suited for this mission. Appendix E is an extract of a report prepared by Eagle Engineering for the Martin Marietta - Michoud Operations. It is included in this report with their permission.

Basically, structural support of an additional payload compartment on the aft y ring of the space shuttle external tank (ET) may permit orbital flight test not possible with the orbiter alone. Utilization of residual propellants in an open-loop turbine drive system may permit mechanical drive of an off-the-shelf 5 MWe generator. Residual propellants may be adequate for 2 hours of steady - state MPD testing at the 5MWe level and longer duration test of smaller units. This interval can be extended by reducing shuttle payload carried and intentionally increasing residual propellants remaining in the ET. There is a trade of about one mass unit of propellant for each unit of payload off-loaded. With full utilization of this potential, it is possible that test periods of 8 hours or more may be provided at 5MWe. Longer duration test of 5GWe thrusters may require a large dedicated flight test spacecraft, possibly a part of the NASA - proposed "Space Operations Center" (SOC).

6. Recommended Future Activities

Recent initiation of research thruster testing at new vacuum chamber facilities at Princeton and JPL has enabled data acquisition on the MPD thruster concept at a much higher rate than has been heretofore possible. As a consequence, the ability to gain empirical information and to validate theoretical models is greatly enhanced over earlier years.

The first recommendation is that this new facility capability be fully utilized, with adequate personnel and support resources to permit facility-limits to be reached at each of the two test installations. To do otherwise will be wasteful of the investment in the facilities.

Second, the available analytic and test information should be utilized to develop a fully ordered test program to explore thruster geometry, power loss deposition, propellant distribution, etc. which will lead to an enhanced ability to configure and produce performance predictions of flight engines for each of the several possible applications. In particular, "down sizing" to the 1MWe or smaller size of the MPD should be explored for the steady-state engine operation, orbit-raising application. The rationale is simply that 1MWe solar arrays will be flown much earlier than 5MWe or larger power supplies.

Third, periodic updating (semi-annually or more frequently) of projected mature flight engine characteristics should be published by JPL and/or Princeton University for use in applications studies.

Fourth, MPD engine subsystem studies should be initiated, with emphasis upon thermal dissipation, mechanical integrity and minimum weight. A significant body of work in active cooling, propellant valve conceptual design, material characterization and engine/power conditioner integration should be anticipated. To guide these efforts, preliminary "model specifications" on flight MPD engines should be written to establish performance goals, life and restart needs, and other requirements. Care should be taken to not overstate the performance and life goals.

Fifth and finally, the MPD flight engine conceptual designs should be employed in comprehensive systems studies. Recommended vehicle configurations for these studies include:

(1) An "IOTV" using a solar array of not more than 1MWe BOL output for orbit transfer from LEO to GEO, using a continuously-powered MPD system.

(2) A 200 to 400 KWe NEP planetary spacecraft with flow rate capability and/or energy storage sufficient to perform efficient "powered fly by" maneuvers at various perigee or perijove altitudes for Delta Vega or Jupiter gravity-assist missions. The MPD thruster would be pulse-powered for this application.

This activity should be organized into a cohesive, time-phased 3 to 5 year development plan, updating the existing MPD development plan.

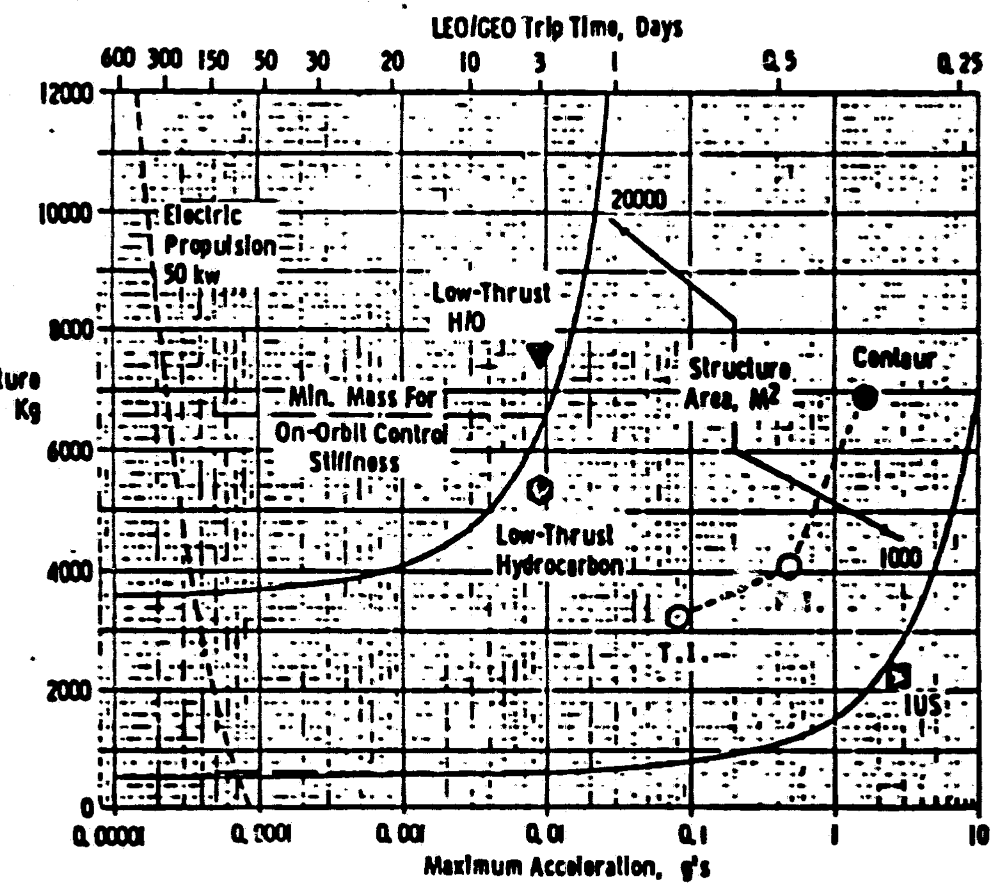
APPROACH

- o REVIEW AVAILABLE LITERATURE, INCLUDING RECENT JPL DATA
- o REVIEW SOLAR POWER SATELLITE ORBIT-RAISING STUDIES
- o CONCEPTUAL SYNTHESIS OF FLIGHT VEHICLE(S)
- o IDENTIFY THRUSTER CHARACTERIZATION NEEDED FOR APPLICATION STUDIES
  - ENGINE DESIGN-RELATED
  - VEHICLE INTEGRATION
  - OPERATIONAL FACTORS
- o IDENTIFY AVAILABLE GROUND TEST THERMAL/VACUUM FACILITIES
- o SYNTHESIZE SHORT-LIFE (~ 10 HRS.) DEVELOPMENT FLIGHT TEST APPROACH
- o CRITIQUE EXISTING DEVELOPMENT PLAN
- o RECOMMEND FUTURE ACTIVITIES

EAGLE ENGINEERING

MAGNETO-PLASMA-DYNAMIC (MPD) THRUSTER APPLICATION

EFFECT OF ACCELERATION ON STRUCTURE WEIGHT - SPACE BASED RADAR MISSION



Ref: Lewis Research Center Presentation - Low Thrust Orbit Transfer, W. Plohr, 1979.

CONCLUSIONS

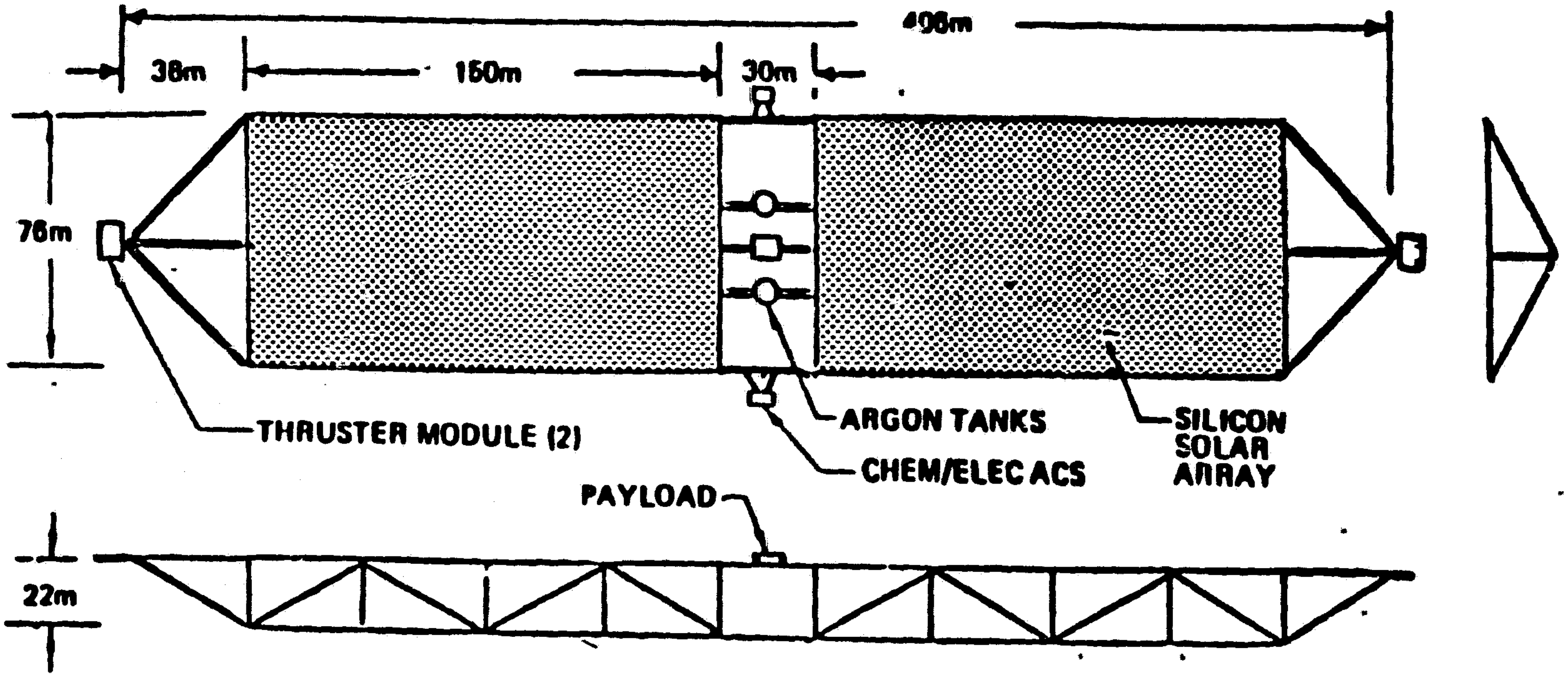
- STRONG INTERACTION BETWEEN ACCELERATION, MASS AND AREA
- ACCELERATION OF ~.01G ENABLES 20-FOLD AREA INCREASE OVER IUS
- IUS RULED OUT
- LOW-THRUST CHEMICAL A STRONG CONTENDER

FUTURE OTV TECHNOLOGY STUDY

FOTVTS-080

ELECTRIC ORBIT TRANSFER VEHICLE
INITIAL POINT DESIGN

LaRC/BOEING



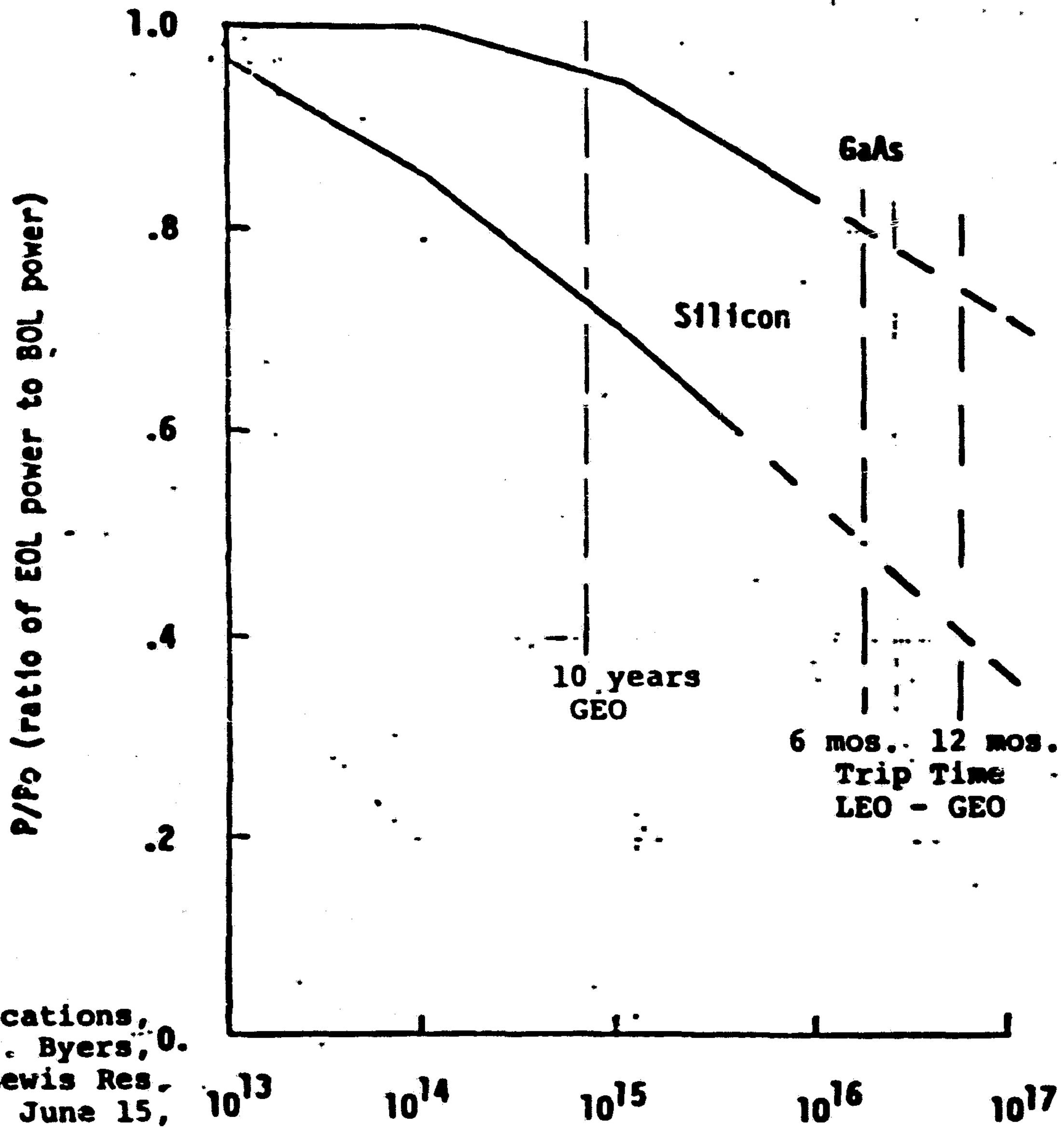
- SPECIFIC IMPULSE = 8000 SEC
- UP TRIP = 180 DAYS
- DOWN TRIP = 69 DAYS
- INITIAL POWER = 4.1 MW
- MAX THRUST = 79N

- PAYLOAD UP = 80 MT
- PAYLOAD DOWN = 0
- FIXED MASS = 40.5 MT
- ARGON MASS = 13.8 MT
- ARRAY AREA = 22,800 m^2
- NO. OF THRUSTERS (50 CM) = 97

EAGLE ENGINEERING

MAGNETO-PLASMA-DYNAMIC (MPD) THRUSTER APPLICATION

SOLAR CELL DEGRADATION VS. RADIATION DOSAGE



Ref: Private Communications, David C. Byers, NASA, Lewis Res. Center, June 15, 1980

OPERATIONAL FACTORS

- o SOLAR OCCULTATION EFFECTS
  - DEPARTURE ORBIT ALTITUDE AND INCLINATION
  - ACCELERATION LEVEL
  - SOLAR DAY OF DEPARTURE
  - ARRAY THERMAL RESPONSE
  - THRUSTER START-UP TIME
  - CHEMICAL PROPULSION ACS MAY DEGRADE $I_{SP} \sim 20\%$, INCREASE TRIP TIME 35%

- o PLUME IMPINGEMENT
  - ARRAY MUST TRACK THE SUN
  - THRUST MUST PASS THROUGH VEHICLE C.G.
  - PLUME CLEARANCE NECESSARY TO PREVENT DAMAGE
  - BOEING SOLUTION - 4 THRUSTER PANELS AT ARRAY CORNERS
  - DESIRED MPD SOLUTION - SINGLE THRUSTER

OPERATIONAL FACTORS (CONT'D.)

- o DAMAGE POTENTIAL AND EFFECTS
  - MICROMETEOROID IMPACT
  - ORBITAL DEBRIS COLLISION
  - CONSTRAINTS ON JETTISON OF SPENT ELEMENTS

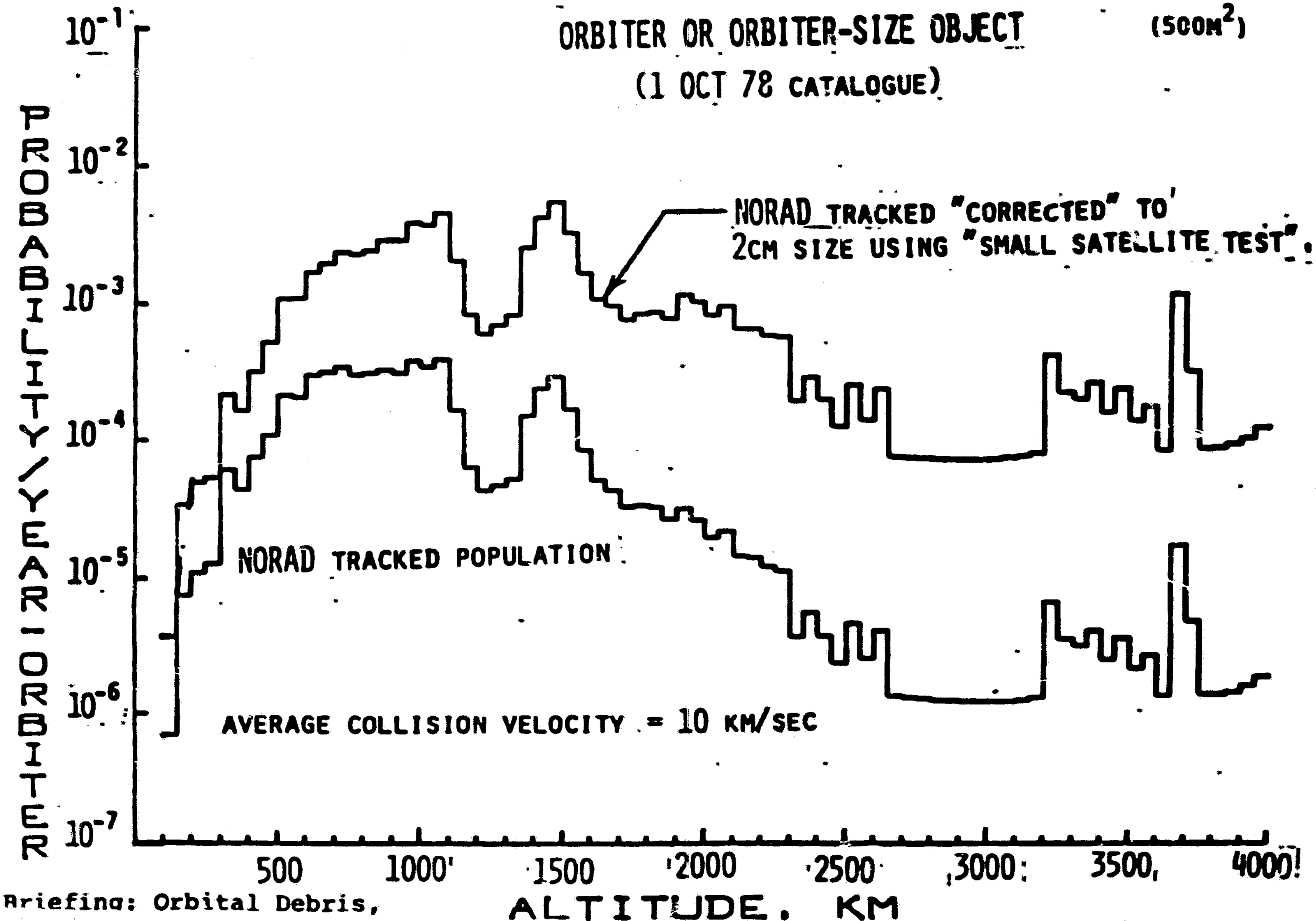
- o MAGNETOSPHERIC INTERACTIONS
  - PLASMA SHORTING OF ARRAY/THRUSTER PANELS
  - DEPOSITION OF EFFLUENT
  - MPD PLUME IONIZATION MAY BE PREFERABLE TO ION ENGINE
  - HYDROGEN MAY BE PREFERRED TO ARGON

EAGLE ENGINEERING

MAGNETO-PLASMA-DYNAMIC (MPD) THRUSTER APPLICATION

PROBABILITY OF COLLISION FOR ORBITER OR ORBITER-SIZE OBJECT (500M²)

(1 OCT 78 CATALOGUE)



Ref: Briefing: Orbital Debris,

1990 AND BEYOND ORBIT RAISING NEEDS

- o LARGE SPACE STRUCTURES WILL REQUIRE:
  - MASS-EFFICIENT ORBIT RAISING

CHEMICAL PROPULSION MASS RATIO		2.68
SPECIFIC IMPULSE	1000 SEC.	1.80
	2000	1.34
	4000	1.16
	8000	1.08

  - ACCEPTABLE OTV INVESTMENT
  - LOW THRUST TRANSFER ($<10^{-2}$ G'S)
  - MODERATE TRIP TIME (< 100 DAYS)
    - MINIMIZE SOLAR ARRAY DEGRADATION
    - MINIMIZE COLLISION POTENTIAL
    - MINIMIZE SYSTEM FAILURES
    - MINIMIZE LOSS OF REVENUE
- o ION ENGINE ORBIT TRANSFER VEHICLE
  - COMPLEX, HEAVY, EXPENSIVE POWER CONDITIONING
  - LOW THRUST/UNIT REQUIRES MANY THRUSTERS/VEHICLES
  - LOW THRUST PROVIDES LONG TRIP TIMES (200-800 DAYS)
  - START LOSSES TBD
  - EXCEEDS SPECIFIC IMPULSE NEEDS

1990 AND BEYOND ORBIT RAISING NEEDS (CONT'D.)

- o MPD ENGINE ORBIT TRANSFER VEHICLE
  - REQUIRES EITHER 75 MWE OR PULSED-POWER APPARATUS
  - DOWN-SIZING POTENTIAL INSUFFICIENTLY UNDERSTOOD
  - COMPARATIVELY UNEXPLORED AS A FLIGHT ENGINE SYSTEM

MPD FLIGHT ENGINE CONSIDERATIONS

- o ENGINE COOLING
  - ANODE COOLING VIA HEAT PIPE/RADIATOR, POWER TO 500 KW
  - CATHODE SELF-COOLED (?)
  - INJECTOR FACE COOLING (LOW $\dot{M}$)
  - PROPELLANT VALVE THERMAL ISOLATION
  - INSULATION INTEGRITY AT ELEVATED TEMPERATURE
- o POWER CONDITIONING INTEGRATION
  - ELIMINATE REQUIREMENT, IF POSSIBLE
  - MINIMIZE EPDS LINE RUNS (DIRECT MOUNT?)
  - MAXIMIZE EFFICIENCY (RADIATOR)
- o THRUST VECTOR CONTROL
  - GIMBAL PPU/THRUSTER ASSEMBLY
  - ROLL CONTROL FOR SINGLE-THRUSTER OTV (?)
  - AVOID IMPINGEMENT ON VEHICLE
- o SYSTEM REDUNDANCY
  - FLIGHT SPARE UNITS
  - CROSS-STRAPPING
  - DEGRADED ARRAY OPERATIONS

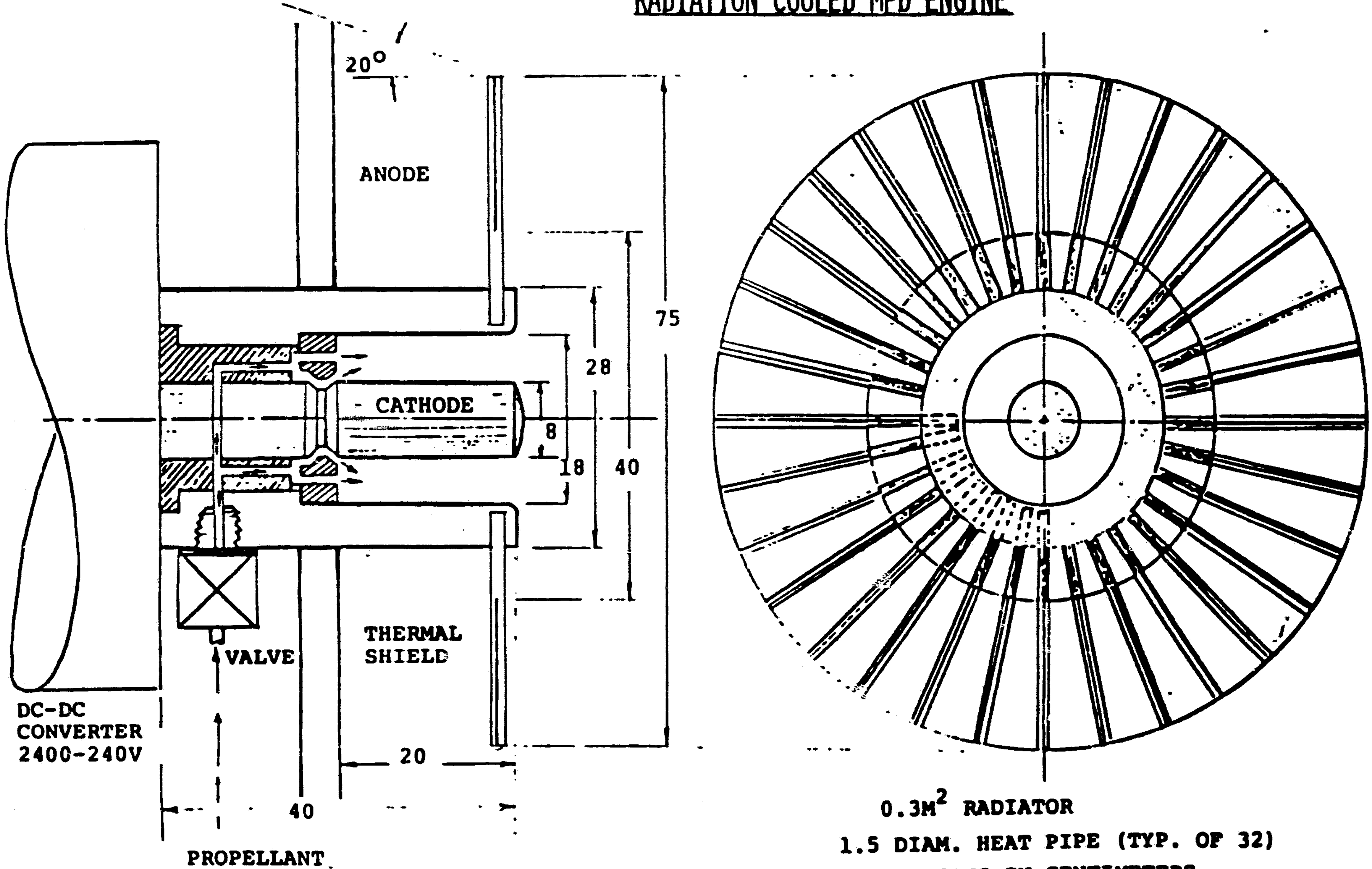
16

EAGLE ENGINEERING

MAGNETO-PLASMA-DYNAMIC (MPD) THRUSTER APPLICATION

CONCEPTUAL CONFIGURATION

RADIATION COOLED MPD ENGINE



0.3M^2 RADIATOR

1.5 DIAM. HEAT PIPE (TYP. OF 32)

VACUUM CHAMBER FOR GROUND TESTS

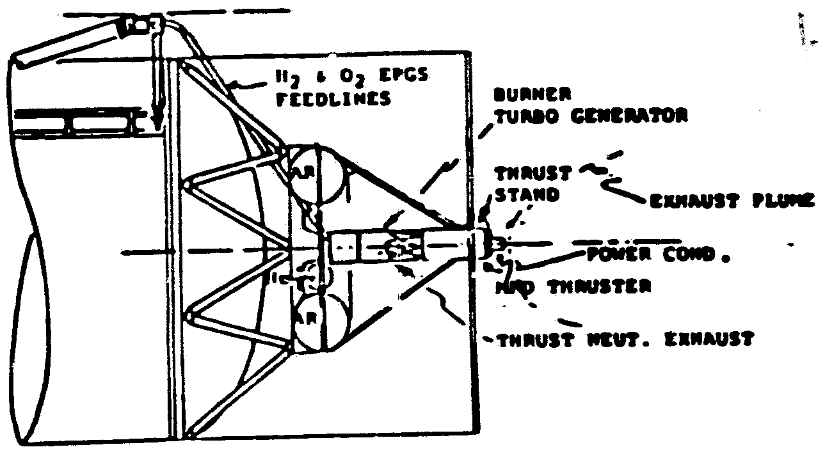
- o TEST REQUIREMENT
  - Electrical Power 5.5 Megawatts
  - Gas Load of 6gm/sec Argon (Equiv. 2500 TL/sec)
  - Test Pressure 10^{-4} Torr
  - Dissipate 5 Megawatts Thermal Energy
- o CHAMBER SURVEY
  - Thirty-one Large Chambers Reviewed
  - None Can Accommodate Gas Load of 6 gm/sec Argon
  - Max Capability Known is 0.12 to 0.24 gm/sec Argon
  - Thermal Energy Will Require Special Test Equipment
- o TWO ACTIVE CANDIDATE CHAMBERS (0.12 TO 0.24 GM/SEC ARGON)
  - NASA JSC CHAMBER A:
    - Working Volume 55' Dia. x 90' Ht.
    - Shroud LIN Cooled to 90K
    - Min Press 10^{-7} Torr
    - Pumping Speed 10^{7} L/Sec
    - Maximum Pulse Duration - 50 MSEC @ 6 gm/sec
- o AEDC, MARK I
  - Working Volume 35' Dia. x 65' Ht.
  - Shroud LIN Cooled to 85K
  - Min Press 10^{-6} Torr

EAGLE ENGINEERING

MAGNETO-PLASMA-DYNAMIC (MPD) THRUSTER APPLICATION

SPACE PROPULSION TECHNOLOGY BENCH

- MOUNT PROPULSION TECHNOLOGY EXPERIMENTS IN ET AFT CARGO COMPARTMENT.
- ALLOW TEST OF ADVANCED PROPULSION CONCEPTS NOT CAPABLE OF SAFE FLIGHT IN SHUTTLE ORBITER PAYLOAD BAY
- ELIMINATES PROVISIONS/CONCERNS OVER ABORTED FLIGHT "HOT" PAYLOAD



CONCEPT OF MPD THRUSTER FLIGHT EXPERIMENT

APPENDIX B

Task

Evaluate two possible solar cell string arrangements for a 2 MW solar array to a steady-state MPD thruster:

(1) Hook up 240 volt sections of the array in parallel and then directly to the MPD, producing 8.3 K amps in the final bus or;

(2) Wire the array in series, producing 2,400 volts DC, 830 amps at the end bus which would then be converted, using a DC to DC converter, to 240 volts, 8.3 K amps.

For a bus of length L, meters and cross section A M^2:

$$R = \text{resistance} = \frac{rL}{A}$$

where r = resistivity (Ω-M). For copper $r = 1.73 \times 10^{-8}$ Ω-M @ $20^{o}C$. For copper, density = 8.9 gms/cm^3.

To compute the power lost, PL, in a given length at bus L, carrying current I, we use

$$PL = I^2R = \frac{I^2rL}{A}$$

We assume the bus will be a thin strip of copper to give it good energy radiating characteristics. To solve the problem of power produced by a given bus length, we must first choose an operating temperature, T, for the copper strip, having radiating area A_R, which will probably be just one side of the strip. We assume the strip radiates as a black body and does not lose energy. in conduction of heat.

$\therefore$ $P_L = A_R \sigma T^4$ for steady-state at temp. T

where $\sigma = .1714 \times 10^{-8}$ BTU/hr-ft^2^{o}R

= the Stetan Boltzman constant

$I\ rL = A_R \sigma T^4$

$$A = \text{cross sectional area} = \frac{I^2 rL}{A_R \sigma T^4}$$

When we choose an operating temp., T for the bus, we must use the resistivity of the given metal at that temp. T. After we compute A, (we guess a value for A_R) we then compare A and A_R to see if they are reasonable for the strip.

We now need a preliminary bus bar layout in terms of length for the 240 volt in parallel scheme. To do this, we need to know what kind of surface area of silicon cells produces 240 volts and what area produces 2,400 volts, so we can lay out a hookup grid for both.

To compare the 2,400 volts to 240V system, we need a weight and efficiency for a DC to DC converter. The radiating area required for this device may be the big number in the whole system.

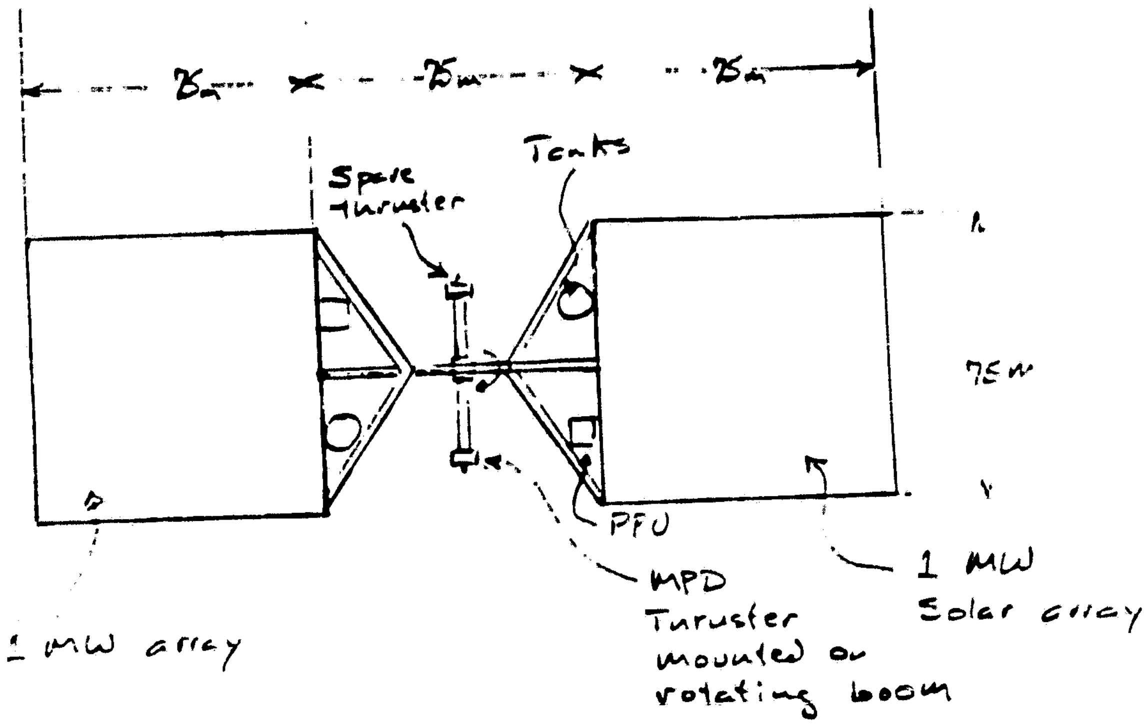
From Augrist, Direct Energy Conversion, 2nd Ed., page 210, we find a 1 cm^2 silicon cell should produce $\approx$0.6 volts and 35 ma or 25 mW. As a check, recall the 1 MW array from the Boeing SPS study was 75m x 75m or $56.25 \times 10^6\ cm^2$ check

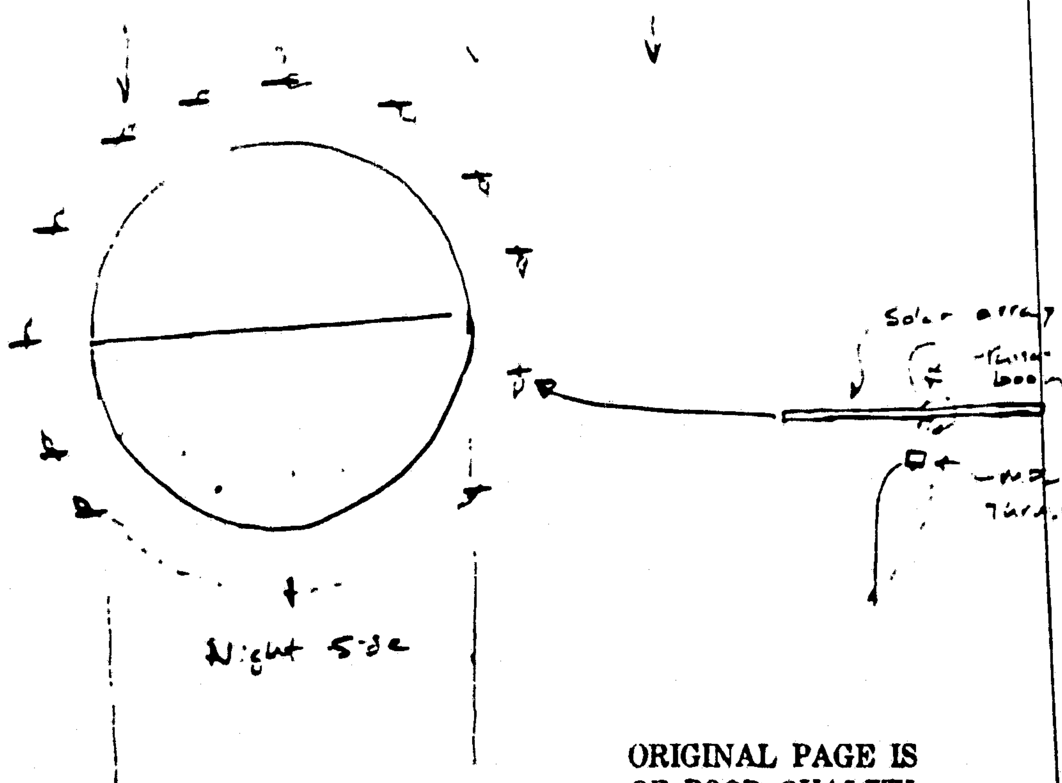
$$25 \times 10^{-3}\ \text{watts/cm}^2 \times 56.25 \times 10^6\ \text{cm}^2$$

$$= 1.4 \times 10^6\ \text{watts}$$

$$\text{or} \approx 1\ \text{MW}$$

2 MW Steady State MPD EOTV





We will assume our 1 cm^2 cells produce .6 V and 30ma or 18 watts/cm^2 so a 75m x 75m array will produce 1 MW.

A 240 volt string would be

$$240 \text{ volts}/.6 \text{ volts/cm} = 400 \text{ cm or } 4\text{m long}$$

A 2,400 volt string would be 40m long.

The diagram on the next page shows a convenient bus bar arrangement for the 240 volt and 2,400 volt spacecraft. The array shape is changed from 75m x 75m to 80 m x 70.3m to allow more convenient spacing.

The upper drawing on the next page shows the 2,400 volt system layout. The strings of 40m x 1cm modules are in two large parallel arrays. Two 2,400 volt bus bars are utilized and one common ground. Each 40m x 1cm string of cells produces 30×10^{-3} amps; therefore every meter of bus bar produces 3 amps; every 20m produces 60 amps. Therefore, each 70m 2,400V bus contributes 210A. The two buses combine to deliver 420A @ 2400V to a DC to DC converter.

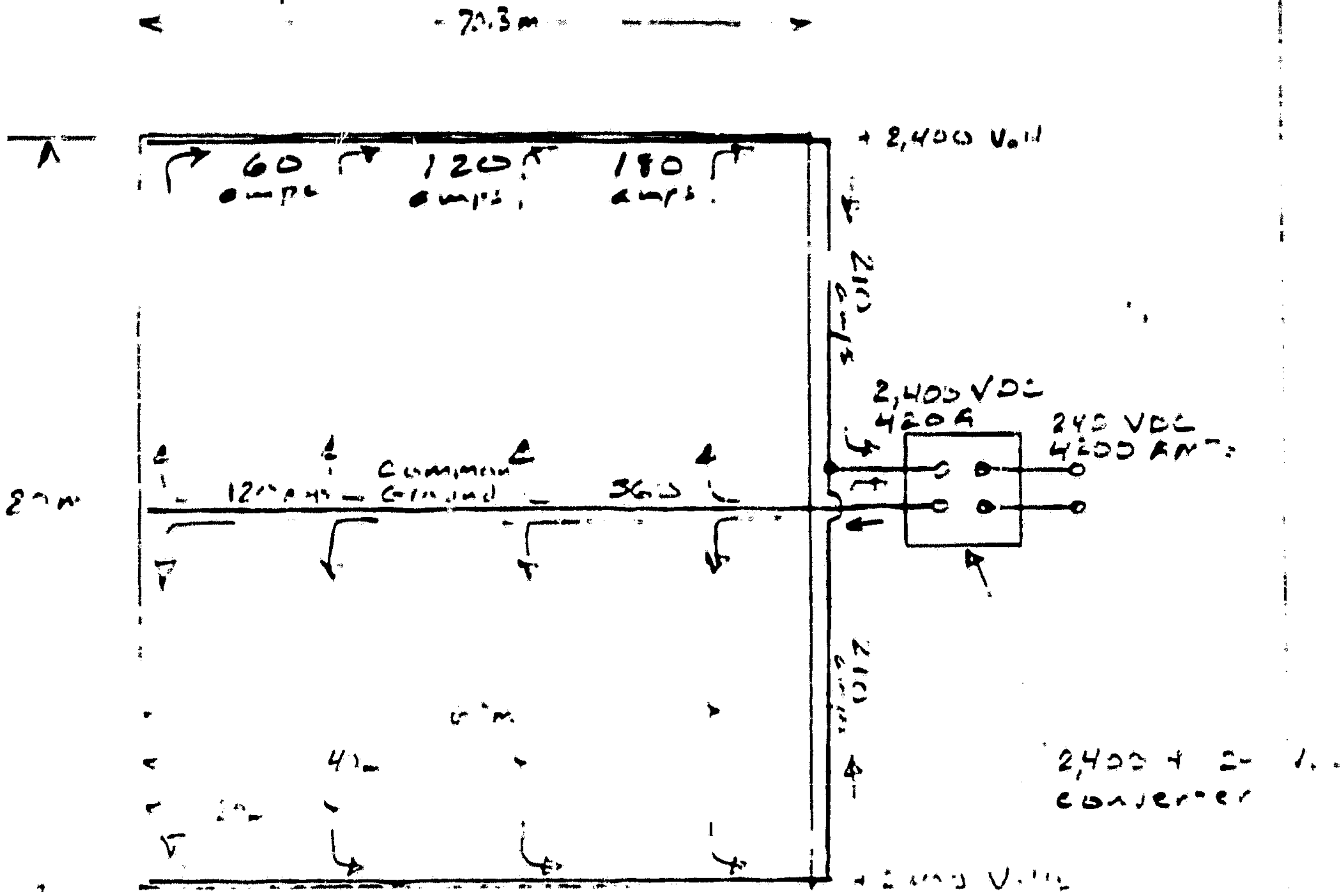
The lower drawing on page 2 details a 240 volt system. A 4m x 1cm string of cells produces 240V @ 30 ma; 70.3 x 100 of these in parallel produces 211 A at 240V, 20 of these hook up to the 240V system producing 4,200 A.

We will compare these two designs on the basis of:

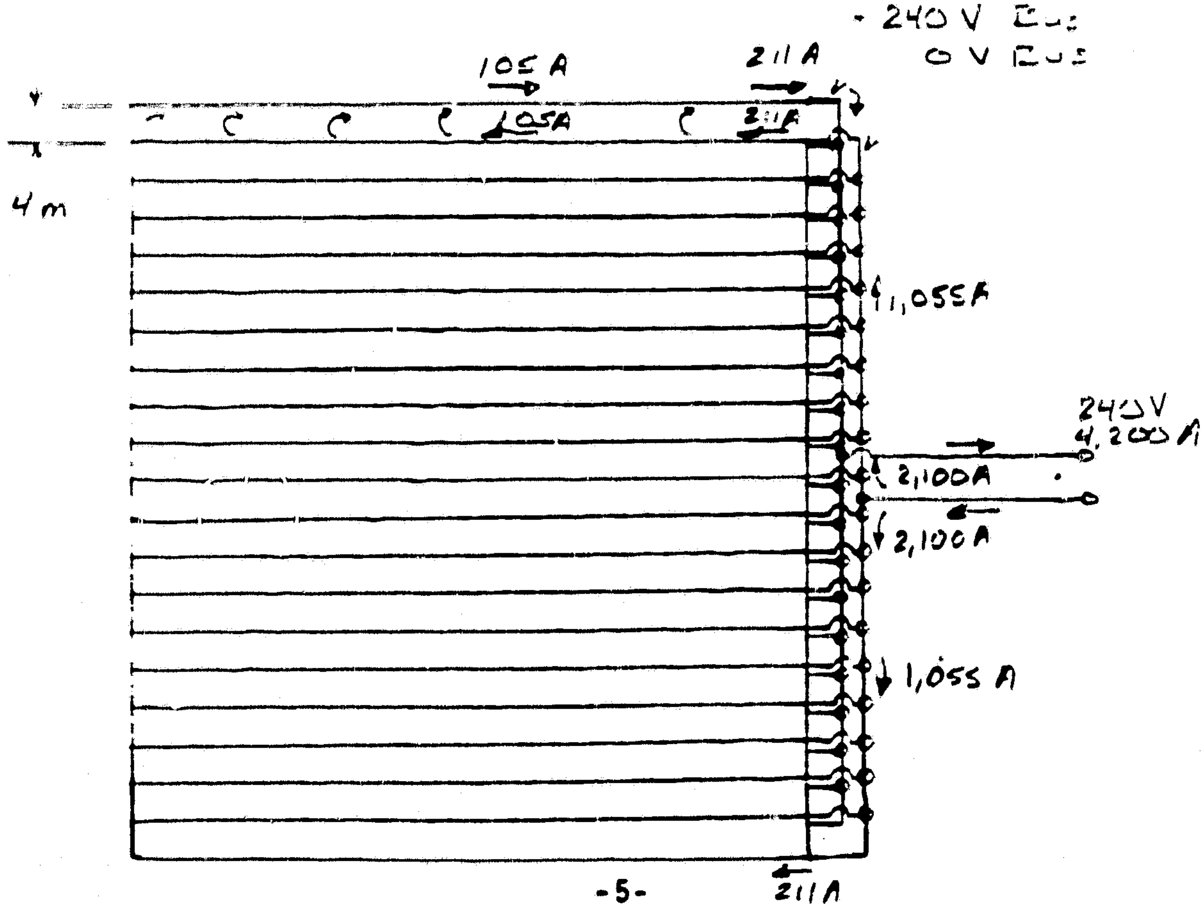
(1) weight

(2) power delivered.

2,400 Volt system, 1 MW Array



240 Volt system, 1 MW



On pages 1 and 2 a method for sizing the wire based on a given operating temp., current, resistivity, and radiating area is discussed. It is not currently clear that this is the best method for optimizing the wire. The optimum wiring layout for a given design will have:

(1) Minimum weight, meaning minimum cross section

(2) Minimum power losses meaning low resistance per unit length, therefore, large cross section and large radiative area to allow a low operating temp.

Resistivity of a length of copper wire increases ≈ 40% for a 100^{o}C temp. rise. The bus bars on the actual vehicle, assuming they serve no structural function (which is another factor to consider) would probably be flat strips of copper tapered to larger cross section as the current conducted increases. Optimization of the wiring appears to be a task somewhat beyond the scope of this study if done in a detailed manner so we will use a crude technique to get wiring weights and power losses.

On the following page, each design is broken into wiring segments, each segment having a given average current to carry. The wire is then sized based on this current and weight and power loss is estimated. The size is chosen by using a table of Allowable Ampacities For Insulated Copper Conductors in the <u>11th Edition of the Standard Handbook for Electrical Engineers</u>. This table is for wiring design in buildings and is not really appropriate to this task, but the technique should give a good rough estimate.

2,400 V, 1MW Array

Segment Current A	Segment Length M	Diameter mm	Weight Kg	Resistance Ω	I^2R W
60	40	4.15	4.7	.051	184
120	60	7.35	22.6	.024	352
180	40	10.40	30.2	.008	259
210	100	12.70	112.6	.014	602
420	30	22.72	108.1	.0013	225
360	20	19.67	54.0	.0013	147
240	20	13.91	27.0	.0023	131
			359.2		1,900

240 V, 1MW Array

Segment Current A	Segment Length M	Diameter mm	Weight Kg	Resistance Ω	I^2R W
105	1,400	6.54	418.6	.72	7,948
211	1,400	11.68	1,334.5	.226	10,063
1,055	80	33.60	630.6	.0015	1,737
2,100	120	38.52	1,243.2	.0018	7,856
			3,626.9		27,604

Since we did not use many segments in this design, the numbers here represent maximum values. In fact, the current assumption (constant max. current in a segment) gives us a power figure that is almost certainly a factor of 2 too high for the 240V case so in comparison we will reduce 27,600 to 15,000W.

The 2,400 volt system requires a DC to DC converter to get the voltage down to 240 volts. This converter might well be a solid state DC to AC inverter, a 10 to 1 transformer to step down and

an AC to DC rectifier circuit on the secondary side. A power loss of 7 to 10 percent each in the inverter, transformer, and rectifier circuit might be expected and would be average values for systems such as this. This would result in an overall efficiency of 75 to 80 percent for the DC to DC converter. It may be possible to do better. The heavy part of this converter would be the transformer core. Just from the wire weights needed to handle these currents, we estimate the converter might weight 300 Kg. The following table gives a mass and power breakdown for the two candidates. Recall also that at 75 percent efficiency with a 2MW array the DC to DC converter must radiate .5 MW requiring a large radiator.

Assume we can run the radiator at $100^oC = 376^oK$

$$P = \sigma A T^4 \qquad A = P/\sigma T^4$$

$$= .5 \times 10^6 \text{ W}/5.7 \times 10^{-12} \text{ W/cm}^2 \times (376)^4$$

$$= 4.38 \times 10^6 \text{ cm}^2$$

20 x 20m radiator

	2,400V, 2 MW	240V, 2 MW
Mass Breakdown:		
Solar cells	*6,000Kg	*6,000Kg
Structure	*1,000Kg	*1,000Kg
Wire	720Kg	7,254Kg
Converter	300Kg	---
Radiator	300Kg	---
	8,320Kg	14,254Kg
Power Losses:		
Wire Loss	3,800W	30,000W
Converter Loss	$.5 \times 10^6$W	---
Output Power	1.5×10^6W	1.97×10^6W
Output Power/Kg	$\boxed{180 \text{ W/Kg}}$	$\boxed{138 \text{ W/Kg}}$

*Taken From Boeing Study Spacecraft.

The watts/Kg numbers indicate the 2,400 volt system is the winner even with the large power losses from the DC to DC converter. We assume that the design with the largest output power per kilogram is the most desirable.

The comparison is sensitive to the converter and radiator weights. An increase, by a factor of 5 for these numbers, would change the 2400V system from 180 W/Kg to 110 W/Kg, making the 240V system at 138 W/Kg the winner. These numbers need to be refined to get a reliable answer. Seventy-five percent for the converter efficiency is a pessimistic number. We can probably do better than this. If the 2400V system wins with 75 percent, it may very well win in the end if a better efficiency is possible. The design does not seem to be sensitive to power losses in the wiring, indicating we might get away with much lower cross section and weight wiring in the 240V system should we iterate both designs again. The wire weight for the 240V system seems to be roughly a factor of 10 greater than that for the 2,400V and is a significant number.

William L. Gill
August 28, 1980

APPENDIX C

PRELIMINARY COMMENTS ON DRAFT OF "GROUND TEST FACILITIES REQUIREMENTS FOR MPD THRUSTER TESTING"

Some characteristics of larger vacuum facilities have been described in Rudy Williams' August Report. A "strawman" configuration of a chamber for MPD testing, Figure 1, can illustrate some of the requirements which may be placed on the selected vacuum facilities. Modification to the thruster to make it compatible with the vacuum facility may have to be considered as well as the converse. The purpose of this note is to suggest particular test demands from which may be derived probable requirements on vacuum facilities and test articles to carry out the test system requirements as described in JPL letter to Eagle Engineering 342/LKR:srh dated August 13, 1980.

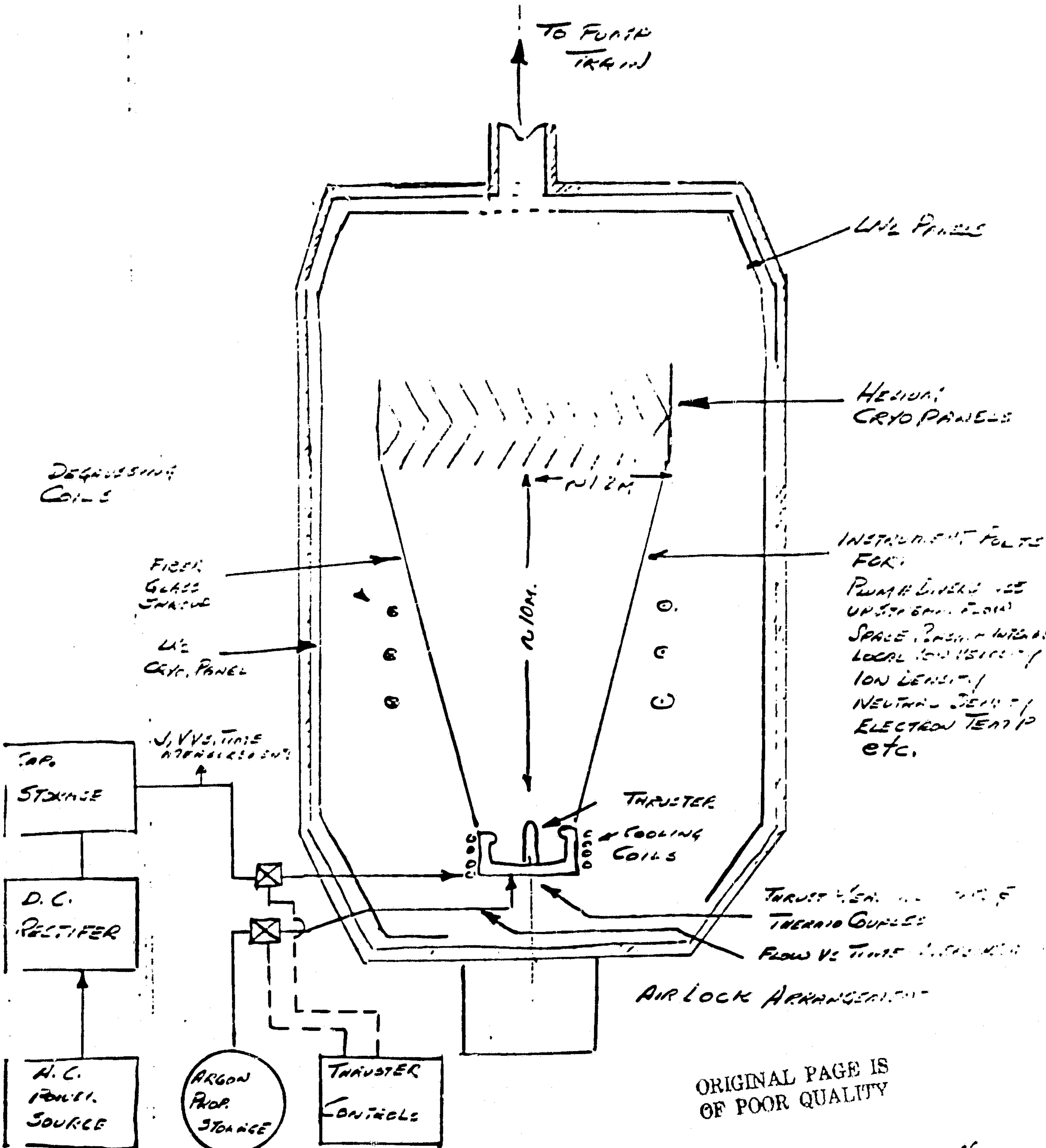
The proposed testing of 100-200Kwe MPD Thrusters requires the dissipation of large amounts of thermal energy and will probably be limited to large test facilities.

The development of a special test article(s) for erosion studies should be considered rather than being a part of performance testing of the thruster. These tests will probably be considerably longer in duration than MPD Thruster performance tests in order to establish erosion rates. Erosion can be dependent upon the angle of incidence of the ion beam, operating temperature, and potential at the eroding surface. In addition, the effects of ions from contaminants produced by sputtering of MPD insulators, electrodes, and other surfaces may require study. It is, therefore, suggested that specialized, small scale test articles which do not require large scale facilities be considered for preliminary erosion tests.

For the systems testing of MPD devices, there are a number of considerations:

FIGURE 1

STRAWMAN MPL TEST ARRANGEMENT



1. Airlock: Will the test article require an airlock for examination or modification of the test article without the requirement to repressurize the main facility?

2. Heat Dissipation: Heat dissipation by the facility is a major consideration, and will affect the facilities vacuum capabilities. The JSC pumping systems in Rudy Williams' summary can dissipate 7Kw (Chamber A) and 1.5Kw (Chamber B), and either would appear, at first glance, to be adequate for the 100-200 Kwe MPD Thruster tests.

The heat to be dissipated will not be uniformly distributed in the vacuum facility. The energy in the MPD plume might range from 20 percent to 50 percent of the input energy to the thruster; and will have to be dissipated so as to minimize reflection, refraction and interaction with the plume. This suggests the need for local cooling capability of from 20 to 100Kwe downstream of the plume. If there is no active cooling of the thruster, the balance of the input energy to the thruster will be dissipated by radiant exchange with the cryo panels near the thruster. The heat to be dissipated would range from 50 to 160Kwe, and could exceed the local capability of the chamber. Active cooling of the thruster from either outside the chamber or by radiators to distribute the heatflux to cryo panels not "seen" by the thruster should be considered. Instrumentation for control and inspection need further definition and can affect the area available for radiator cooling.

3. Field Effects: Chamber sections have been used with fiber glass existing MPD thrust vacuum facilities to minimize interaction with chamber walls. For the testing proposed, the use of degaussing coils in the chamber to cancel the earth's magnetic field may have to be considered. The volume over which the

"zero" field and the allowable field strength need definition, which depends upon the detailed requirements of the plume tests. Since plume shape will probably depend upon Isp, a maximum Isp should be specified to help define degaussing requirements.

4. <u>Vacuum Requirements</u>: The JPL test requirements set an upper limit of 10^{-4} Torr for the pressure in the vacuum facility. Except for the erosion testing, the duration of the ground test is as yet undefined. Considering the large quantity of heat to be dissipated, the temperature of the Argon being removed from the chamber appears variable and uncertain, hence the pumping speed and the time constant for the vacuum facility are uncertain. Requirements should be established on the allowable change in the chamber pressure level while the test is in progress.

5. <u>Power Requirements</u>: The power levels required to operate a nominal MPD Thruster at 100-200Kw may require facility operation at offpeak hours if this power is obtained from a local utility. Both convenience and cost may indicate that onsite generation is desirable and may be necessary. In any event, this power will most probably be delivered to the vacuum facility as AC, since the operating voltage of the thruster is low compared to typical transmission lines. Details of capacitor charging/discharging will influence the design of the rectifiers used for conversion to DC.

6. <u>General</u>: The testing of MPD Thrusters in ground vacuum facilities will be limited by both vacuum and heat dissipation. If work is limited to the configurations stated in JPL's system test requirements, it would appear that the 100-200Kwe operation can be carried out in some existing test facilities, but the 5.3Mwe tests do not appear feasible. Significant down sizing of the thruster should be investigated further to permit

continuous operation, or to permit high pulse repetition rates which more closely reproduce steady state operation. The test problem is analagous to the mission application - both need smaller power demand MPD Thrusters.

7. Thruster System Testing: The JPL test requirement draft document indicates that performance and lifetime qualifications should be carried out under actual system operation for propellant handling, injection, energy storage, switching and thermal control subsequence. Considering all the constraints on chambers which are imposed by thruster testing, it would appear that such subsystem testing should be carried out separately. "All up" systems tests are necessarily deferred to space flight testing.

Franklin U. (Rudy) Williams
August 29, 1980

APPENDIX D

VACUUM CHAMBER SURVEY

A compilation of large thermal vacuum chambers is submitted in accordance with the request to survey test facilities that could be candidates for ground tests of the MPD Thruster. References used to complete the survey are as follows:

1. NHB 8800.5 (NASA) - Technical Facilities Catalog.
2. NASA CR-1876 Vol III 1971 - Inventory of Aeronautical Ground Research Facilities.
3. TN-D-1673 - Survey of Large Space Chambers.
4. Condensed From ARTC 6477 - Catalog of Large Space Chambers (Pressure Less Than 10^{-4} Torr).
5. MSC-03415 - Test Facilities of E&D Directorate.
6. SETDM 1001 (NASA-JSC) - Space Environment Test Division Facilities Users Guide.
7. Various Other Company Facility Brochures.

The information presented is approximate dimensions, type of pumping system, time to working pressure, minimum pressure, shroud temperature range, solar characteristics, miscellaneous information (remarks), and status of the facility (active, deactivated, or unknown).

During the next report period, ground test requirements for the MPD Thruster will be compared to these large thermal vacuum chamber's capabilities to determine the suitable chamber(s). Assessment will be based on factors such as pumping speeds, electrical power availability, and thermal heat dissipation capabilities. The objective is to locate a thermal vacuum ground test facility that is capable of testing the MPD Thruster at approximately 6gm/sec. argon while maintaining pressures below 10^{-4} torr if possible.

Otherwise, the maximum operating characteristics of the MPD Thruster will be described with respect to the chamber(s) most nearly satisfying the 6gm/sec flow rate (other requirements being satisfied). It is noted that flow rates on the order of (0.12 to 0.24 gm/sec. argon) can be accommodated by some of the large facilities.

SOME LARGE THERMAL VACUUM FACILITIES

1	CHAMBER LOCATION	Chamber A NASA/JSC, Houston	Chamber B NASA/JSC, Houston	Space Environ. Sim. NASA/GSFC, Greenbelt	25-FT Space Sim. NASA/JPL, Pasadena	Mark 1 Aerospace Chamber USAF/AEDC, Tullahoma
2	OUTSIDE DIMENSIONS FT.	65 dia. X 120 Ht.	35 dia. X 43 Ht.	35 dia. X 60 Ht.	27 dia. X 90 Ht.	42 dia. X 82 Ht.
3	WORKING VOLUME FT.	Vacuum: 55D X 90H Top Solar: 130 Beam Max. Side Solar: 13W X 33H Beam Max.	Vacuum: 25D X 26H Top Sun: 13D Beam	Vacuum: 27.5D X 40H Top Sun: 20W X 17.5L Beam	Vacuum: 25D X 70H Top Sun: 19D X 25H Beam	Vacuum: 35D X 65H Top Sun: 15W X 20L Beam
4	PUMPING SYSTEM	DP: Valved & Trapped Cryo: 17-20K Helium 7kw capacity	DP: Valved & trapped Cryo: 17-20K Hel. 1.75kw capacity	DP: Trapped Cryo: 15K Helium	DP: Valved & Trapped Cryo: None	DP: Valved & Trapped Cryo: 20K Helium
5	PUMP DOWN TIME HRS.	7	5	13	3	12
6	MINIMUM PRESSURE TORR	$\sim 10^{-7}$	$\sim 10^{-7}$	$\sim 10^{-8}$	$\sim 10^{-6}$	$\sim 10^{-6}$
7	SHROUD	Size: Working Volume Range: 90-310K	Size: Working Volume Range: 90-300K	Size: Working Volume Range: 90-425K	Size: Working Volume Range: 90-400K	Size: Working Volume Range: 85-300K
8	SOLAR SIMULATOR	Type: On-Axis Carbon ARC Modular	Type: On-Axis Xenon Filtered Modular	Type: On-Axis H_g-Xenon Modular	Type: Off-Axis Xenon Variable Beam and intensity not filtered	Type: On-Axis Sun Gun Tungsten iodide
9	REMARKS	Man rating with full capability of life support and monitor rotating floor	Man rating with full capability of life support and monitor	Spacecraft Positioner	9' Beam-970W/Ft^2 11' Beam-590 W/Ft^2 15' Beam-280 W/Ft^2 19' Beam-200 W/Ft^2	LH_e Traps For Propulsion Tests
10	STATUS	Active	Active	Active	Active	Active

SOME LARGE THERMAL VACUUM FACILITIES

1	CHAMBER LOCATION	Electric Propulsion 25' NASA/LeRC, Cleveland	Spacecraft Propulsion Res. NASA/LeRC, Sandusky	Space Power Facility NASA/LeRC, Sandusky	Cryo-Propellant Research NASA/LeRC, Sandusky	Chamber A Boeing, Kent
2	OUTSIDE DIMENSIONS FT.	25 dia. X 70 long	42 dia. X 70 Ht.	100 dia. X 122 Ht.	25 dia. sphere	39 dia. X 49 Ht.
3	WORKING VOLUME FT.	Vacuum: 20D X 60L	Vacuum: 33D X 55H	Vacuum: 80D X 80H Top Sun: 15W X 30L Beam	Vacuum: 20D Sphere	Vacuum: 29D X42H Top Sun: 7D X 30H
4	PUMPING SYSTEM	DP: Baffled Cryo: None	DP: Valved & Trapped Cryo: None	DP: Valved & Trapped Cryo: None	DP: Valved & Trapped Cryo: None	Ion & Sublimation Cryo: 15K Helium
5	PUMP DOWN TIME HRS.	24	24	12	3	8
6	MINIMUM PRESSURE TORR	$\sim 10^{-7}$	$\sim 10^{-7}$	$\sim 10^{-6}$	$\sim 10^{-6}$	$\sim 10^{-8}$
7	SHROUD	Size: Working Volume Range: 90-300K	Size: Working Volume Range: 80-300K	Size: 40D X 40H Removable Cyl. Wall Range: 90-300K	Size: Working Volume Range: 80-300K	Size: Working Volume Range: 85-300K
8	SOLAR SIMULATOR	None	None	Type: 400kw Argon Module Top Sun: 1 Module	None	Type: On-Axis Xenon Filtered Modular
9	REMARKS	Chamber maintained at vacuum - test articles transferred in-out.	Rocket firing at altitude	Equipped and shielded to handle radioactive test articles	Cryogenic ullage tank testing	
10	STATUS	Active	Deactivated	Deactivated	Deactivated	Active

SOME LARGE THERMAL VACUUM FACILITIES

1	CHAMBER LOCATION	Solar-Thermal-Vacuum Chamber GE, Valley Forge	Thermal Vacuum Chamber GE, Valley Forge	TRW, Redondo Beach	TRW, Redondo Beach	Martin-Marietta, Denver
2	OUTSIDE DIMENSIONS FT.	32 dia. X 54 Ht.	39 dia. sphere (2)	22 dia. X 64 Ht.	30 dia. sphere	29 dia. X 65 Ht.
3	WORKING VOLUME FT.	Vacuum: 21D X 30H Top Sun: 15D X 30H Beam	Vacuum: 21D X 25H	Vacuum: 20D X 35H Top Sun: 10W X 10L Beam	Vacuum: 27D sphere Side Sun: 8W X 8H Beam	Vacuum: 22D X 56H Top Sun: 16D X 16H
4	PUMPING SYSTEM	DP: Valved & Trapped Cryo: 20k Helium	DP: Trapped Cryo: 15K Helium	DP: Valved Cryo: None	DP: Valved Cryo: None	DP: Valved & Trapped Cryo: 20K Future installation
5	PUMP DOWN TIME HRS.	8	6	5	5	6
6	MINIMUM PRESSURE TORR	$\sim 10^{-8}$	$\sim 10^{-8}$	$\sim 10^{-6}$	$\sim 10^{-6}$	$\sim 10^{-8}$
7	SHROUD	Size: 21 Ft. Cylinder Range: 100-300K	Size: Working Volume Range: 85-300K	Size: Working Volume Range: 80-475K	Size: Working Volume Range: 240-375K	Size: Working Volume Range: 100-300K
8	SOLAR SIMULATOR	Type: Off-Axis Xenon Not Filtered	None	Type: On-Axis Xenon 30kW	Type: On-Axis Xenon	Type: Off-Axis Xenon Filtered
9	REMARKS		150kW IR Heating Capacity			
10	STATUS	Active	Active	Active	Active	Active

SOME LARGE THERMAL VACUUM FACILITIES

1	CHAMBER LOCATION	McDonnel-Douglas, Huntington Beach	McDonnel-Douglas, Huntington Beach	The.mo Vacuum Chamber RCA, Princeton	Dynamic Lab Langley	Propulsion Lab Lewis
2	OUTSIDE DIMENSIONS FT.	39 dia. sphere	30 dia. X 36L	30 dia. X 34 Ht.	55 dia. X 55 Ht.	15 dia. X 63L
3	WORKING VOLUME FT.	Vacuum: 21D X 25H	Vacuum: 28D X 30L	Vacuum: 24D X 20H Volume	Vacuum: 55D X 55H	Vacuum: 13D X 50L
4	PUMPING SYSTEM	DP: Trapped Cryo: 20k He	DP: Trapped & Valved Cryo: None	DP: Trapped Cryo: None	DP:	DP:
5	PUMP DOWN TIME HRS.	8	8	8	12	17
6	MINIMUM PRESSURE TORR	$\sim 10^{-8}$	$\sim 10^{-7}$	$\sim 10^{-7}$	10^{-4}	10^{-7}
7	SHROUD	Size: Working Volume Range: 100-300K	Size: Working Volume Range: 100-400K Shroud Removed	80-400K Shroud	Ambient	Ambient
8	SOLAR SIMULATOR	7' X 9' SS 2-30 kW Xenon portable modules	No	None	None	None
9	REMARKS				Side 20 x 20	End 15D
10	STATUS	Active	Deactivated	Active	Unknown	Active

SOME LARGE THERMAL VACUUM FACILITIES

1	CHAMBER LOCATION	Arc Melting Furnace Lewis	Space Power #1 Lewis	Tank #5 Lewis	HEAO MSFC --	AEDC Tullahoma
2	OUTSIDE DIMENSIONS FT.	20x20x20	30 dia. X 100L	15 dia. X 60L	20 dia. X 25L	19 dia. X 32L
3	WORKING VOLUME FT.		Domed Sect. 20dia. X 100L			18 dia. X 29L
4	PUMPING SYSTEM	DP:	DP:	DP:	DP:	DP:
5	PUMP DOWN TIME HRS.	4	30	19	--	6
6	MINIMUM PRESSURE TORR	10^{-6}	10^{-6}	10^{-6}	10^{-7}	10^{-6}
7	SHROUD		80-350K	80-350K	80-	80-300K
8	SOLAR SIMULATOR		None	None	None	None
9	REMARKS		End 20D	End 15D	End	End 19D
10	STATUS	Active	Unknown	Unknown	Active	Active

SOME LARGE THERMAL VACUUM FACILITIES

	CHAMBER / LOCATION	ARFPL Edwards, Calif.	Bendix Ann Arbor	Grumman Bethpage Space Simulation	Hughes El Segundo, Calif.	Propulsion Lab Wright-Patterson AFB
2	OUTSIDE DIMENSIONS FT.	30 dia. sphere	20 dia. X 27L	19 dia. X 28Ht.	15 dia. X 36 Ht.	--
3	WORKING VOLUME FT.	28 dia. sphere	18 dia. X 20L	15 dia. X 20Ht.	14 dia. X 35H	23 dia. X 27H
4	PUMPING SYSTEM	DP:	DP:	DP:	DP:	DP:
5	PUMP DOWN TIME HRS.	--	12	6	2	8
6	MINIMUM PRESSURE TORR	10^{-6}	10^{-7}	10^{-7}	10^{-6}	10^{-6}
7	SHROUD	90-475K	90-425K	80-425K	80-410K	80-300K
8	SOLAR SIMULATOR	None	Yes	None	Yes	None
9	REMARKS	Top	End 20D	Top 15D	Bottom 15D	--
10	STATUS	Unknown	Unknown	Unknown	Active	Unknown

SOME LARGE THERMAL VACUUM FACILITIES

1	CHAMBER LOCATION	18' Chamber McDonnell-Douglas, St. Louis, Missouri				
2	OUTSIDE DIMENSIONS FT.	18 dia. X 30L				
3	WORKING VOLUME FT.	15 dia. X 25L				
4	PUMPING SYSTEM	DP:				
5	PUMP DOWN TIME HRS.	12				
6	MINIMUM PRESSURE TORR	10^{-7}				
7	SHROUD	100-300K				
8	SOLAR SIMULATOR	None				
9	REMARKS	End 18D				
10	STATUS	Unknown				

APPENDIX E

SPACE SHUTTLE

EXTERNAL TANK

ON ORBIT MISSIONS

with an

AFT CARGO COMPARTMENT

MARTIN-MARIETTA CORPORATION

CONTRACT #ASO-707242

August, 1980

Prepared by:

Eagle Engineering, Inc.
17629 El Camino Real
Suite 125
Houston, TX 77058

INTRODUCTION

The present Space Transportation System has defined vehicle capabilities which in turn, establish limits on the users in terms of maximum mass, payload volume, payload design envelopes, mission stay-time, etc. Taking these "top-level" requirements, the users have been further restricted by other vehicle limits ranging from utilities (power, thermal control, data recording, data transmission) to dynamic load factors, vibration levels, and material selection for use in design. In addition, operations such as venting, pointing, shroud deployment, radiated energy, etc. have limited experimenters in design and operations. It is NASA's job to establish "compatible" manifests of payloads to insure that combined payloads will stay within the capabilities of the STS system, that the individual experiments will not interfere with each other, and insofar as possible, the best combinations of payloads are selected to meet launch windows, KSC ground flow, etc. All of these considerations have created some uncertainty and reservation in the payload community which to a large part tends to hinder attracting certain classes of payloads.

On the other hand, the greater-than-anticipated success of the "Getaway Specials" (GAS), the small, self-contained payloads, has created a demand that NASA is attempting to solve by considering a rack for multiple canister attachments instead of the present longeron single-canister attachment system.

For instance, there are some 330 GAS payloads that have submitted earnest money, and the potential seems to be limited only by the availability of flights.

Therefore, it appears that NASA's marketing of the STS is already limited by the capability or restrictions of the STS and not by the potential users in the marketplace.

The thrust of a new study to examine how the STS can increase its capability to deliver more payloads-per-flight or extend on-orbit operations capability leads one to look seriously at the use of an aft compartment on the external tank for the following reasons:

- Provides an increase in diameter - up to approximately 25 feet - as an alternative to the orbiter 15 foot diameter payload bay constraint, thereby reducing packaging complexity and allowing consideration of configurations unacceptable to current orbiter payload bay limits.

- Provides a volume for additional hardware needed to extend on-orbit stay-time or additional utilities for orbiter or payload use.

- Provides a payload volume that relaxes constraints on payloads or experimenters for payload bay materials or contamination control as well as for "hazardous" operations or operations that create manifesting problems or payload bay limitations on venting of hazardous fluids, etc.

- Provides capability for additional flexibility in manifesting cargos and opening earlier launches to GAS customers.

The use of the aft compartment on the external tank requires carrying the external tank to orbit; performance penalties for this mission have been determined in an earlier

study for on-orbit inspection of the insulation protection system on the ET.[1] In addition, the total mass of the STS is increased by the payload mass plus the additional mass of the support structure, shroud, fittings, insulation, etc. These impacts need to be assessed in terms of benefits versus impacts in a systematic fashion using the most current performance and cost estimates available from the STS program.

The accommodation of an aft compartment into the external tank has been evaluated by Martin Marietta and others(2), (3 and 4). However, the wide range of potential users of this available volume also yields a wide range of scar weights and interfaces.

Several types of payloads utilize the increased dimensions available in the aft compartment to allow larger, less complex packaging of deployable antenna, structures for assembly or test, or more efficient design of upper stages. Also, of particular importance, would be the ability to carry cryogenic propulsive stages in this volume that are infeasible to carry in the orbiter payload bay because of the hazards and costs associated with the propellants and resulting complex propellant handling systems. Indeed, the NASA planetary mission hardware could be greatly increased in size and capability if the higher propulsive capabilities of LOX/LH_2 could be used for transfer stages.

(1) MMC, "Conceptual Study of an Orbital Inspection of the Thermal Protection System (TPS) on the External Tank", July 1980.

(2) IAF Preprint IAF80-A41, "The External Tank as a Large Space Structure Construction Base", N. J. Witek and T. C. Taylor.

(3) AAS80-089, "Commercial Operations for the External Tank in Orbit", T. C. Taylor

(4) IAF Preprint IAF80-1AA46, "Global Benefits of the Space Enterprise Facility Using the External Tank", N. J. Witek and T. C. Taylor.

The report presents the results of the review, by Eagle Engineering, of mission types and payloads to identify the potential benefits (and impacts) afforded by the use of the aft cargo compartment, together with a rational set of screening criteria to allow classification and prioritization of these missions for future study.

Based on the original Microfiche, multiple pages appear to be missing from this document

TITLE: Alternate Location For Cryogenic Upper Stages

CATEGORY: 4(c)

CONCEPT DESCRIPTION: Utilize area aft of external tank for mounting the proposed cryogenic stage (Orbit Transfer Vehicle, "OTV"); study sharing of propellants between Shuttle abort flight and cryogenic stage to enhance useful mass to orbit.

MISSION REQUIREMENTS:

NOMINAL MISSION: Launch spacecraft destined for high orbit in Shuttle payload bay; OTV in ET aft cargo compartment. Insert to 90x90NM orbit, assemble upper stage to spacecraft with aid of Shuttle RMS, augmented as necessary.

ABORTED MISSION: At abort decision, utilize OTV propellants for orbiter MPS to aid performance, jettison OTV along with ET during abort sequence.

BENEFITS:

1. Orbiter does not have to install complex cryogenic stage provisions.
2. Aborted missions do not have to accomplish OTV propellant dump and safing for landing. Orbiter always lands light.
3. Enhanced flight safety of STS with cryogenic stage.
4. Potential of enhanced performance of Shuttle by utilizing OTV propellants for abort (O/F shift and FPR?).
5. Potentially lighter weight OTV and support cradle by improved mounting and (trade study) launch without O_2 load in OTV - load during ascent from ET. ($\approx$3 to 4% of ET LO_2).
6. Relieves OTV design constraints imposed by orbiter.
7. Full 60' of orbiter payload bay available for long payloads.

STS IMPACTS: (1) Requires development of external tank aft cargo compartment (ACC), OTV provisions for ACC, penetration of ET LH_2 and LO_2 feedlines near orbiter quick disconnects, (2) Requires placing ET into low (90NM?) orbit and subsequent de-orbit with attendant performance penalties. (3) Requires development of OTV and payload on-orbit assembly equipment and procedures. (4) OTV mounting accommodations expended with ET on each mission. (5) OTV lost on aborted STS missions. (6) Ascent flight abort procedures will be complicated if OTV propellants are depended upon for abort, FPR's, O/F shift, etc.

REFERENCES:

1. Centaur Systems Familiarization GDC/LVP79-012, 1 March 1979, General Dynamics,Convair Division.
2. Centaur in Space Shuttle for Launch of Galileo Mission 30 April 1979, General Dynamics,Convair Division.
3. STS/Centaur Safety Characteristics, 21 August 1979, General Dynamics, Convair Division.
4. OTV Concept Definition Study NAS8-33533, First Quarterly Progress Review, 2 October 1979, General Dynamics, Convair Division.
5. OTV Concept Definition Study NAS8-33533, Final Study Review, 8 July 1980, General Dynamics, Convair Division
6. OTV Concept Definition Study NAS8-33532, Final Briefing - Task 1 Mission Analysis July 1980, Boeing Aerospace.
7. Private Communications
   (a) Vince Calouri - Boeing
   (b) Humboldt Mandell - Johnson Space Center
8. Orbital Propellant Handling and Storage Systems Definition Study GDC/ASP-79-002, Final Report Volume I - Executive Summary, 15 August 1979, General Dynamics, Convair Division.
9. Orbital Propellant Handling and Storage Systems Definition Study GDC/ASP-79-002, Final Report Volume II - Technical, August 1979, General Dynamics, Convair Division.

ORBITAL TRANSFER VEHICLE

FOR

EXTERNAL TANK AFT CARGO COMPARTMENT

$W_p \geq 30,000$ lbm O_2/H_2

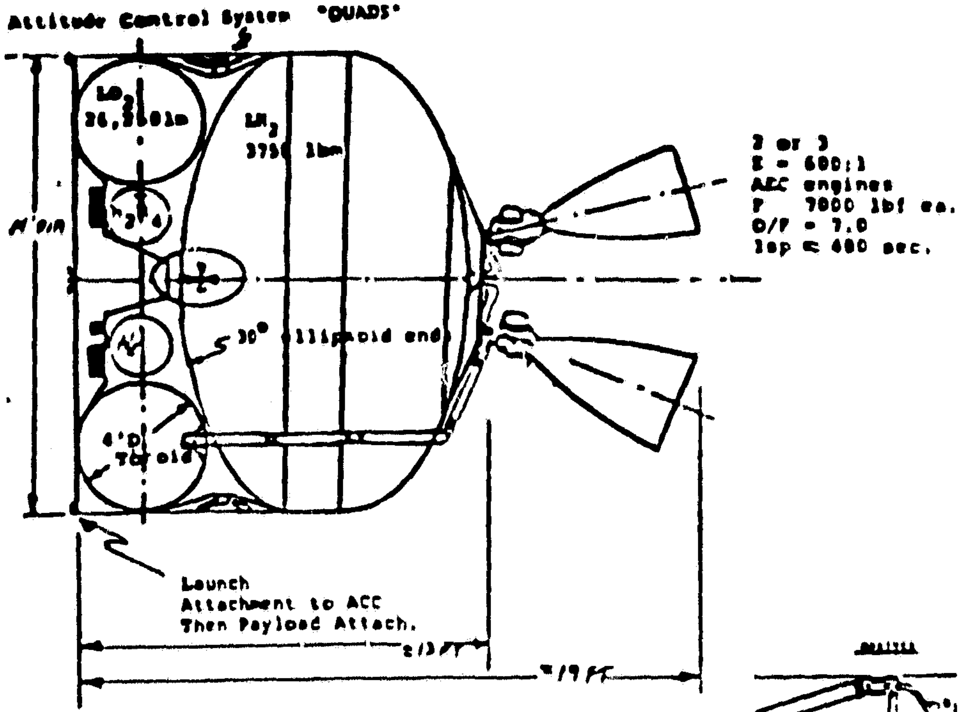


FIGURE I

One conceptual configuration of an Orbital Transfer Vehicle (OTV) with a cryogenic main propulsion system, attitude control system and avionics.

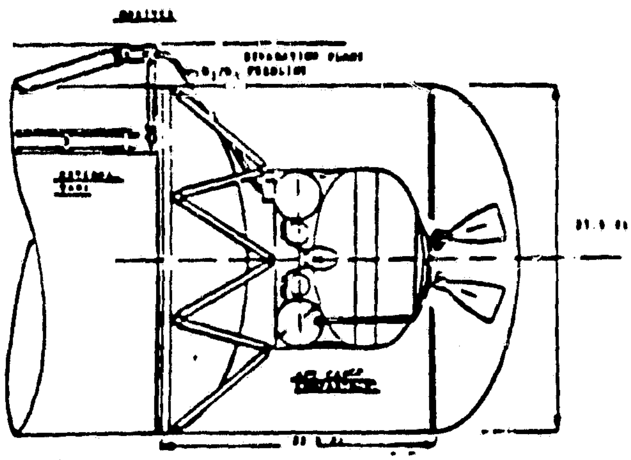


FIGURE II

The OTV mounted in the aft cargo bay compartment similar to the Apollo lunar excersion module mounting.

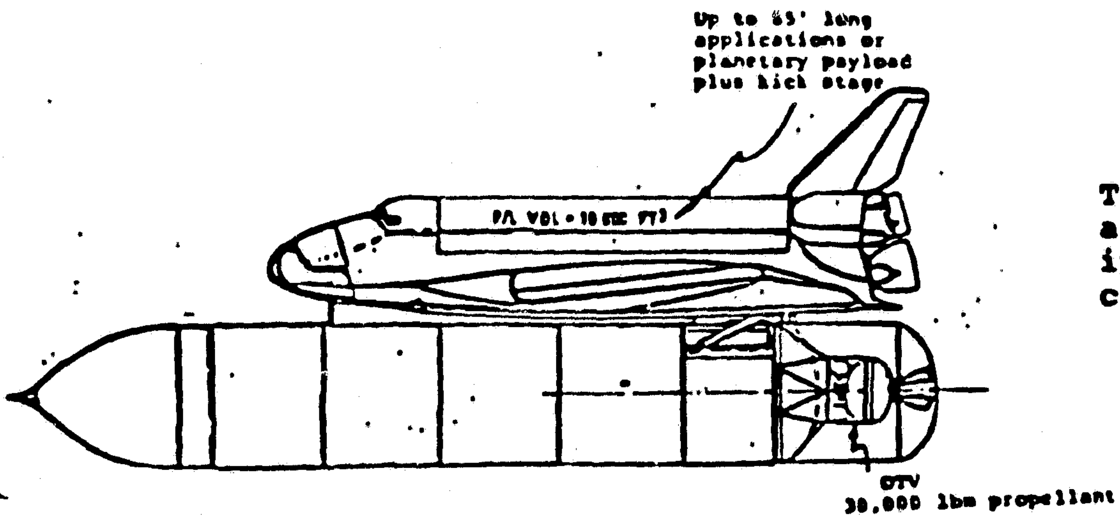
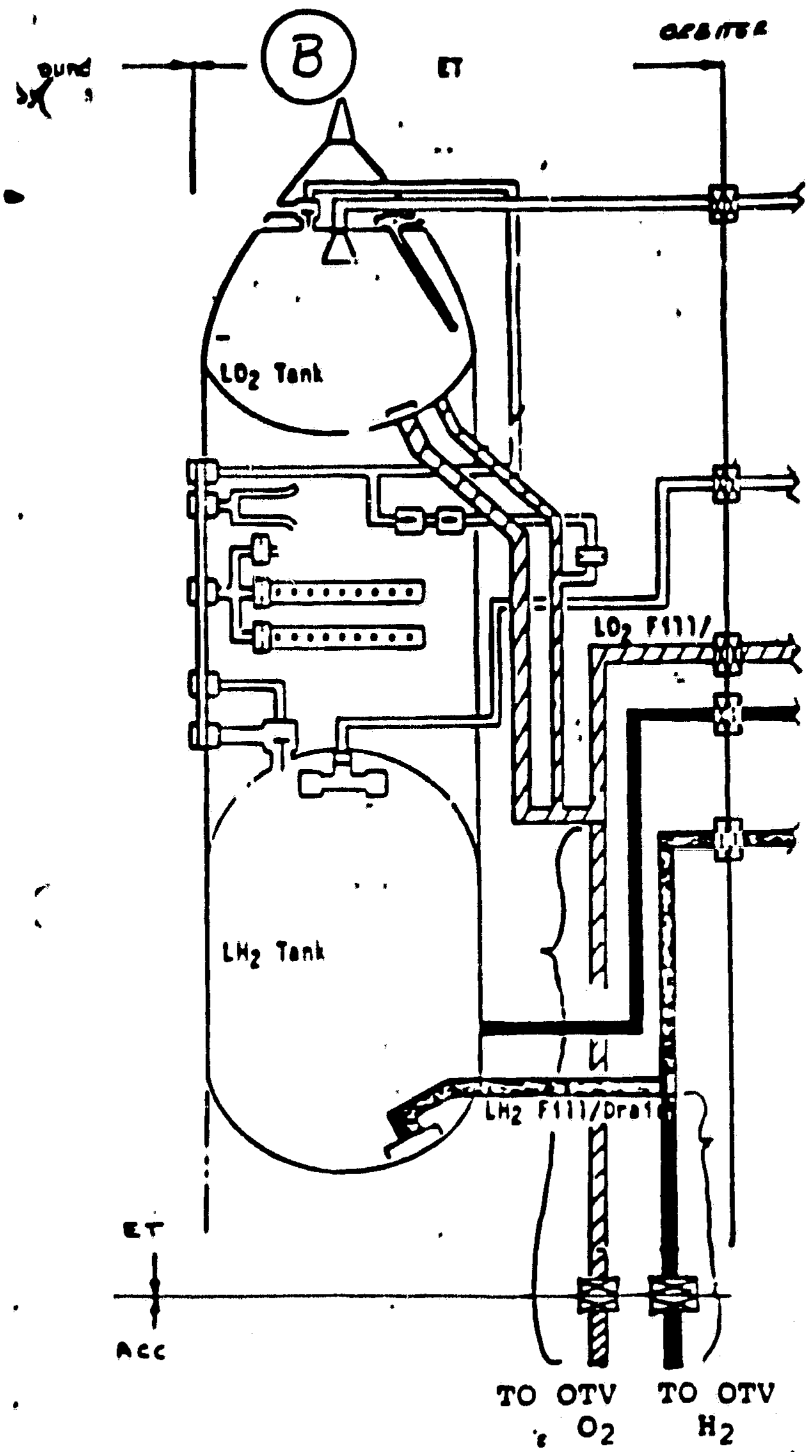
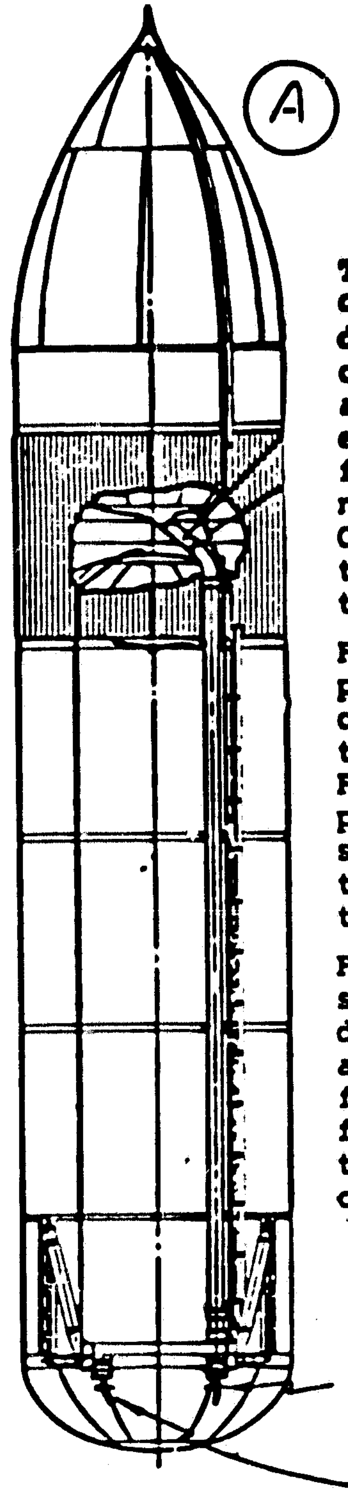


FIGURE III

The overall orbiter and ET with OTV in the aft cargo compartment.

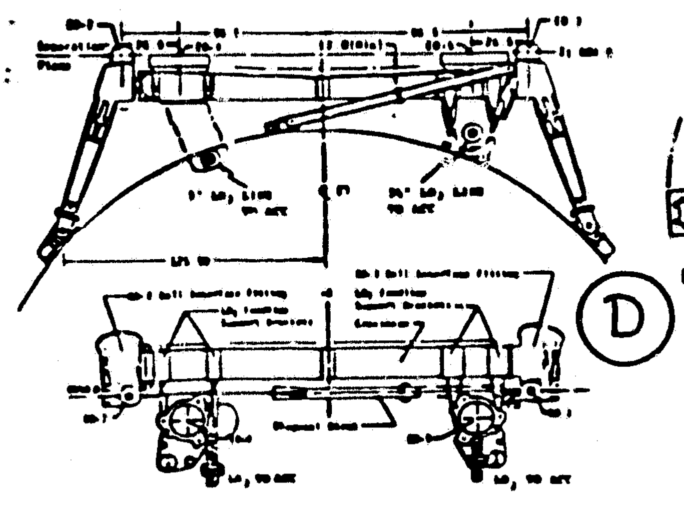


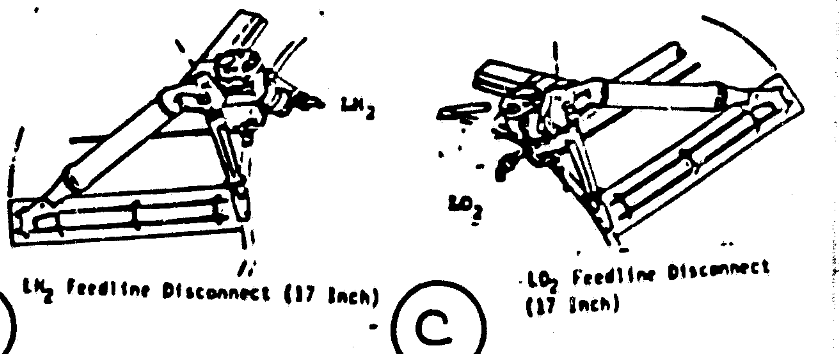


This composite group of illustrations depict one concept of routing the O_2 and H_2 from the external tank fill/drain lines near the ET-to-Orbiter disconnects to the O_2 and H_2 tanks in the ACC.

Fig. "A" shows the physical location of the lines relative to the O_2 while Fig. "B" schematic provides a more specific illustration of the connection points.

Figures "C" & "D" shows in greater detail where the attachments interface with the feedlines and ET-to-Orbiter disconnect.





TITLE: Short-Term, High-Power Electrical Test Bed

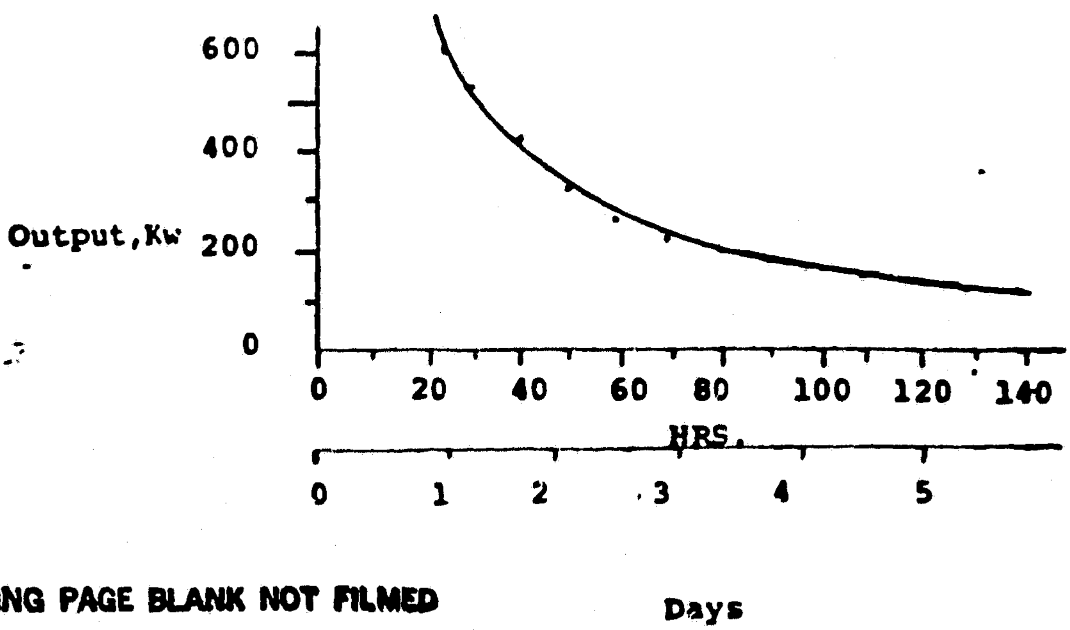
CATEGORY: 4(b)

OBJECTIVE: Several proposed space projects require a source of high level electrical power. These projects include (a) microwave power beaming experiments; (b) high power radar; (c) arcjet propulsion systems; and (d) military defensive systems.

CONCEPT DESCRIPTION: The external tank, placed in orbit with its contents of residual hydrogen and oxygen supply constitutes a valuable fuel supply. This fuel can be used to produce electrical power on orbit.

Two means are proposed to be investigated to generate electrical power utilizing the fuel; they are: (a) moderate power levels (100's of kilowatts) using fuel cells and (b) high power levels (in megawatts) by use of an open loop gas turbine driven generator.

Assuming ET fuel residuals of a little over 13,000 lbs., the electrical power output versus time for a fuel cell will be approximately as follows:



PRECEDING PAGE BLANK NOT FILMED

Duration of Power

16-19

The weight for the fuel cell and radiator are (TBD).

A concept for the open loop gas turbine is provided in the following sketch (Figure 1):

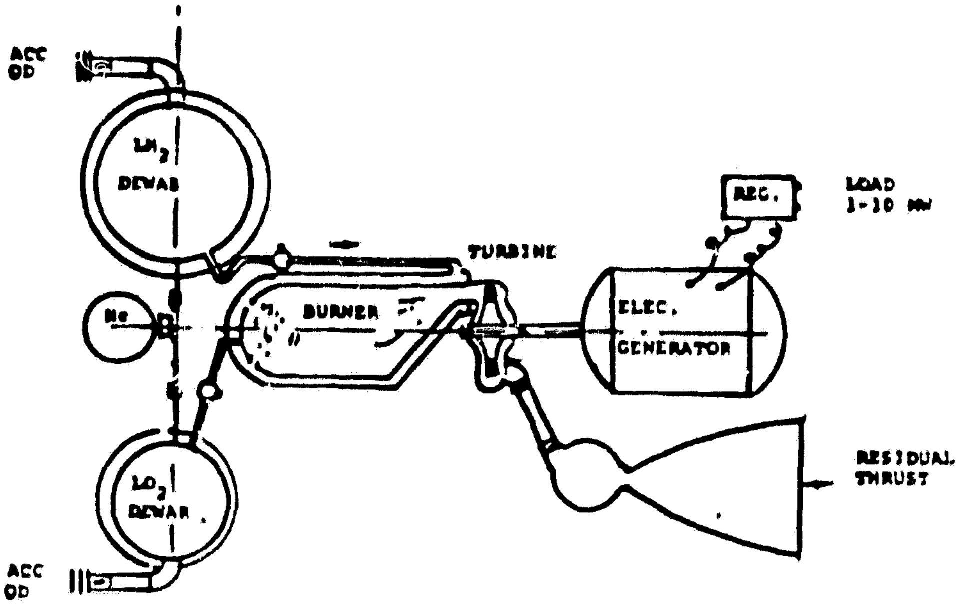


FIGURE 1

A conceptual installation of the turbo generator in the aft cargo compartment and connection to a power utilization device located in the orbiter payload bay is shown in Figure II:

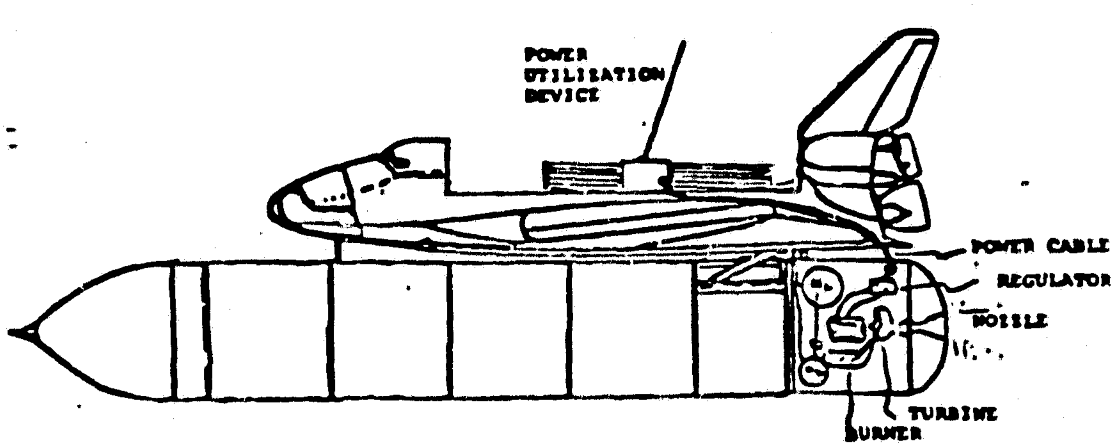


FIGURE II

The anticipated electrical power output vs. time is shown in the table below:

POWER LEVEL (MEGAWATTS)	TIME (HOURS)
1	10
2	5
3	3.3
4	2.5
5	2.0
6	1.67
-	
-	
-	
10	1.0

MISSION REQUIREMENTS:

The on orbit stay-time is a function of the specific power utilization device to be flown. The orbit altitude is directly related to stay-time, but it is anticipated that an altitude on the order of 120NM will be adequate to cover the maximum operating time of the fuel cell of 7 days. Shorter missions on the order of 2-3 days may be as low as 90 NM.

BENEFITS:

This mission and payload will permit the accomplishment of experimentation beyond the current Shuttle vehicle capability. In addition, the electrical power generating system is located remotely to the orbiter reducing mission hazards in the event of an abort.

STS IMPACTS:

(1) Power Generator operating controls and safety systems will be required in the orbiter crew compartment.

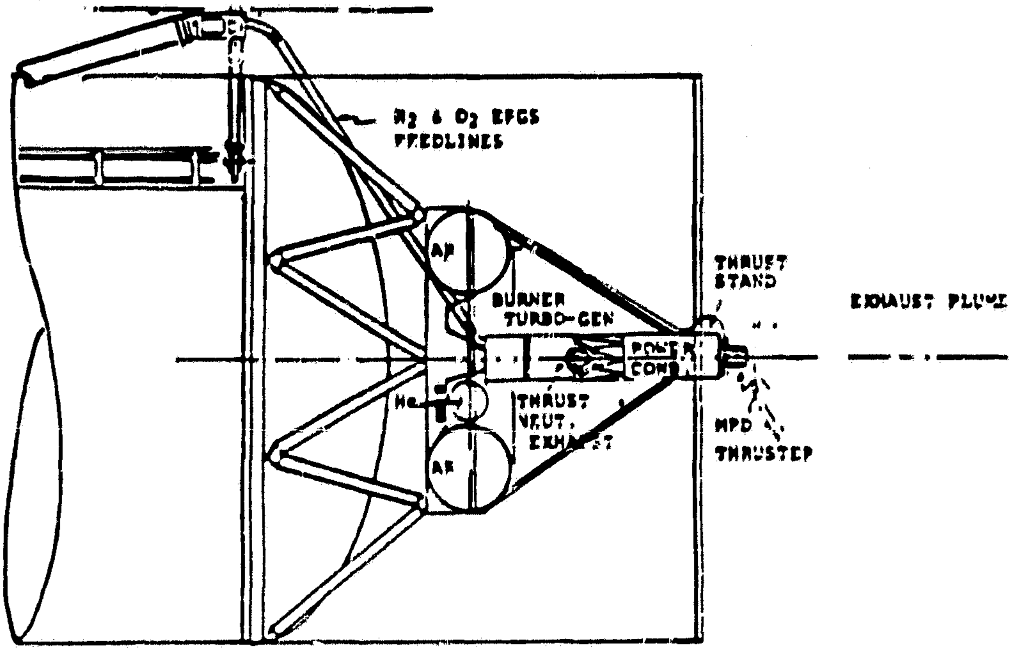
(2) Power cable routing to the payload bay power utilization may be incorporated in the orbiter/ET interface or routed externally by EVA.

(3) Requires development of mission procedures.

TITLE: **Space Propulsion Technology Bench**

CATEGORY: 4(c)

CONCEPT DESCRIPTION: Mount propulsion technology experiments in aft cargo compartment of ET which are not acceptable for ascent flight or use on orbit in Shuttle orbiter payload bay. See Figure 1.



CONCEPT OF MPD THRUSTER FLIGHT EXPERIMENT
FIGURE 1

MISSION REQUIREMENTS: N/A

BENEFITS:

1. Allow test of advanced propulsion concepts not capable of safe flight in Shuttle orbiter payload bay.
2. Eliminates provisions/concerns over aborted flight and landing with "hot" payload.
3. Provides better field-of-view for potentially contaminating exhaust plumes.
4. Permits utilization of thrust produced for orbit makeup.
5. Increased orbital vehicle inertia may enhance test operations.

24 - 33

STS IMPACTS:

1. Requires new thinking toward safety in ET ACC region.
2. Loss of test article unless retrieved by EVA and/or manipulator.

REFERENCES: Eagle Engineering, Inc. - In-house Concept.